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SOME LIMITING FACTORS OF CHEMICAL ROCKET MOTORS

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SUMMARY

The factors limiting the possible performance of rocket motors using chemical propellants are discussed. In particular, the effects of varying propellant choice, combustion chamber pressure, and nozzle expansion ratio, are investigated in some detail, with presentation of numerical results.

It is suggested that some increase in expansion ratio, as compared with present practice, is desirable even for satellite rockets taking-off from the Earth's surface, while for rockets operating all the time in space, a considerable increase will be necessary. Higher expansion ratios will, in general, encourage the use of higher chamber pressures, in order to reduce overall bulk and weight of the motor.

Cooling considerations are very much influenced by the above factors and will probably limit permissible chamber temperatures to about 3,500° C., or perhaps 5,000° C. with refractory materials and the use of techniques for sweat cooling and reduction of chamber wall stresses. Permissible temperatures between these values will in future allow the use of more energetic propellants than those now in common use, and such a trend appears essential for the practical realization of even the easier interplanetary missions. Propellant density will have an important influence on tankage weight.

It is concluded that the more ambitious interplanetary missions (e.g., a return lunar voyage starting from, and returning to, the Earth's surface, or a return Mars voyage from an initial orbit outside the atmosphere) are likely to remain forever beyond the capabilities of the chemical rocket, whatever propellants are selected, unless very extravagant ratios of take-off to payload mass can be tolerated.

Introduction

The prospect of space flight has captured the imagination and retained the interest of a great number of people, many of whom were first attracted by its technical aspects. Among the reasons for this must be included the fact that so many of its problems as yet remain unsolved. Both at present and for some time to come, these problems will provide a stimulating challenge taxing the ingenuity and resourcefulness of workers in a broad range of scientific fields. Among them, we can quite safely include the design and development of the means by which the space vehicle will derive its motive power, since the practicability, if not the possibility, of any venture beyond the atmosphere will depend most of all on the means of propulsion available.

Ten years ago, except perhaps in the minds of those with the most fertile imaginations, the chemical rocket motor was the only practical means worthy of consideration. Within the last decade, however, the release of nuclear

energy has been adequately demonstrated. The opportunity has quickly been taken to speculate upon the ways and means by which this new source of power could be utilized to promote flight into space, Ref. 1 being a notable example. It is perhaps because of, rather than in spite of, the advent of this competitive form of prime mover, that a review of some of the possible capabilities of the chemical rocket is not out of place at this moment. That is in fact what this paper attempts to do, indicating the extent to which improvements in performance might be expected, and relating the resultant problems to those which have already been encountered and overcome.

To begin with, it must be made quite clear that a *complete* assessment of the chemical rocket's capabilities is too complex and detailed a study for the confines of this short paper. Consequently, only broad and general principles can be discussed and some of the more subtle implications of, for example, the use of higher chamber pressures and temperatures are left unexplored.

Again, it must be understood that no assessment of the effect of the rocket motor's performance, on the capabilities of the vehicle it propels, will give completely the right answer if the study is confined only to the rocket motor. For example, the choice of propellant combination will largely determine the exhaust velocity obtainable. But at the same time, the mean specific gravity of the propellants will also decide the volume of the tanks required, which will affect their weight regardless of the actual amount of propellant carried. Thus will the empty weight of the vehicle be affected, and this will of course critically influence its ultimate performance. The tankage weight will also depend to some extent on the particular duty which the vehicle has to perform, and this might influence, in some way, whether a "light" or "heavy" propellant combination be used. The comparison of a vehicle which is to be accelerated away from the Earth's surface, and one which is only to be accelerated out of an orbit into deeper space, is an example of this. In the former case, the tanks will be necessarily heavier in order to withstand the high acceleration required to minimize air resistance and gravitational losses. In the latter case, however, accelerations and inertial forces can be kept low and the tanks can be of comparably lighter construction.

The above comparison can also be used to illustrate another factor. Any improvement in the efficiency of the rocket motor, by the use of a higher combustion pressure or expansion ratio, will probably result in an increase in its specific weight (i.e., weight per unit of thrust generated). A higher specific weight will have a much more serious effect when the thrust of the motor is large compared with the weight of the vehicle, as in the case of take-off from the Earth's surface, because the motor weight then constitutes a more considerable fraction of the empty weight of the vehicle. In the case of acceleration away from an orbit, however, the motor weight will be a comparatively small item, so that a high specific weight could be tolerated, with the attendant benefit of improved thermodynamic performance.

These examples are cited merely to indicate that the ultimate potentialities of the chemical rocket motor cannot be completely assessed unless some parallel consideration is given to the duty and design of the particular vehicle of which it is the power plant.

Typical Missions

In order to provide some sort of yardstick by means of which the usefulness of the chemical rocket can be measured, three possible expeditions into space have been considered. They are, in effect, expeditions for which the chemical rocket might have been expected to be a suitable propulsive unit. The three cases have been chosen primarily because they represent voyages of progressively increasing difficulty, in the chronological order in which they are likely eventually to be achieved.

The cases considered are (*a*) a take-off from the Earth's surface and the acquisition of sufficient velocity to permit a stable orbit round the Earth, (*b*) acceleration away from this orbit, followed by a landing on the Moon and a return to the orbit, (*c*) acceleration away from the same orbit, leading to a landing on Mars and once again a return to the orbit. In each case the characteristic velocities assumed are those given in Ref. 2, while that assumed for the last journey is actually about the same as that for an Earth-Moon-Earth journey, if the effect of a small amount of atmospheric braking is included. In each case, calculations have been carried out for a range of structural factors ϵ, this parameter being as defined and used in Refs. 3 and 4. Different numbers of separate steps were considered for each mission, constant values of ϵ being assumed for each step, resulting in optimum values of G, the payload factor. Refs. 5 and 6 discuss the implications of such assumptions and describe the method of calculation used.

Before considering the results in more detail, let us consider some typical values of exhaust velocity and structural factor which have already been obtained. Nitric acid, hydrogen peroxide and liquid oxygen, burned respectively with aniline, "C stoff" and an ethanol/water mixture are fairly representative proven propellant combinations and these give exhaust velocities of 2·25, 2·2 and 2·35 km./sec. respectively. In addition to these fuels, kerosene or petrol has been also used with each of the above oxidants, the exhaust velocities being then 2·3, 2·3 and 2·5 km./sec. respectively. These exhaust velocities assume expansion ratios currently used (c. 20 : 1), average combustion and expansion losses, and have been corrected for operation at low or zero ambient pressures. With regard to structural factor, V2, WAC Corporal and Viking are probably typical of current practice and these have values of $\epsilon = 0{\cdot}25$, 0·44 and 0·21 respectively. A figure of 2·5 km./sec. thus appears to be about the best exhaust velocity presently obtainable, while a value of $\epsilon = 0{\cdot}20$ (based on existing military and research rockets) represents the present limit of structural efficiency.

Bearing in mind these values, let us consider the three missions one by one. In each case it has been assumed that they would be quite impracticable if it required an initial weight of more than 500 tons to carry 1 ton of payload, and undesirable if this were greater than 200 tons. For the earth satellite project, it appears then, that this would be marginal using chemical propellants already in general use, regardless of the structural efficiencies obtained or the number of steps used. If, however, the exhaust velocity were increased to about 4 km./sec., the project would be clearly practical if a three step rocket were used.

In fact, it would not then be worthwhile to use more steps than three, unless it were found impossible to obtain structural efficiencies as good as $\epsilon = 0{\cdot}2$.

For the second case, that is, the lunar journey, the use of exhaust velocities of the order of 2 to $2\frac{1}{2}$ km./sec. again leads to an impractical answer. If the exhaust velocity were increased to 4 km./sec., the operation would be feasible if a structural factor of 0·2 and five steps were used. If exhaust velocities of the order of 6 km./sec. were possible to achieve, three steps would suffice, so long as a structural factor no worse than 0·2 was obtainable.

The conclusions to be drawn from a study of the third mission (the Mars journey) are very clearly defined. Unless structural factors better than 0·1 can be obtained, the use of exhaust velocities of the order of 2·0–2·5 km./sec. admit to no solution whatsoever. An increase of exhaust velocity to 4 km./sec. hardly alters the picture unless one can tolerate an overall mass ratio in excess of 6,000 and then only if a structural factor of 0·1 and the use of as many as ten separate steps is assumed. An increase of exhaust velocity to 6 km./sec. admits a possible solution, just within the practical limits which have been arbitrarily assumed.

Summarizing very briefly, therefore, it appears that exhaust velocities of the order of 4 km./sec. are necessary to make even the least difficult mission really practical, and that structural factors better than 0·2 may even then be mandatory. Let us now examine the prospects for achieving such an advance, assuming that the chemical rocket motor is used as the method of propulsion.

Thermodynamics

The exhaust velocity, which is in effect a measure of the efficacy of the motor itself, is a function of a number of dependent and independent factors. If the expansion of the gases in the rocket nozzle takes place down to a pressure equal to the ambient value, the exhaust velocity is given by the equation

$$v_e = \sqrt{\frac{2\gamma}{\gamma - 1} \cdot \frac{G\theta}{M} \left\{1 - \left(\frac{P_e}{P_o}\right)^{\frac{\gamma - 1}{\gamma}}\right\}} \quad .. \quad .. \quad .. \quad .. \quad (1)$$

where (in consistent units)

v_e = exhaust velocity, and
γ = ratio of the specific heats of the products of combustion (averaged over the expansion cycle);
G = universal gas constant;
θ = combustion temperature;
M = mean molecular wt. of the gases (averaged over the expansion cycle);
P_e = the ambient pressure;
P_o = the pressure at which the propellants are burned.

If the gases are expanded to a pressure P_x differing from the ambient pressure, an additional term must be added to the above expression to allow for the pressure thrust which must be added to the momentum thrust, and then v_e' becomes the "effective" exhaust velocity given by the equation:

$$v_e' = \sqrt{\frac{2\gamma}{\gamma - 1} \cdot \frac{G\theta}{M} \left\{1 - \left(\frac{P_x}{P_o}\right)^{\frac{\gamma - 1}{\gamma}}\right\}} + \frac{(P_x - P_e)A_x}{dm/dt} \quad .. \quad (2)$$

where A_x = exit area of nozzle and dm/dt is the rate of propellant flow, again in consistent units.

In free space where $P_a = 0$ or at extremely high altitudes when ambient pressure can be neglected, this equation can be rewritten to:

$$v_e' = \sqrt{\frac{2\gamma}{\gamma - 1} \cdot \frac{G\theta}{M} \left\{1 - \left(\frac{P_x}{P_o}\right)^{\frac{\gamma-1}{\gamma}}\right\}} + \frac{P_x A_x}{dm/dt} \quad .. \quad .. \quad .. \quad (3)$$

which can be converted into the form

$$v_e' = \left[\sqrt{\frac{2\gamma}{\gamma-1}\left\{1-\left(\frac{P_x}{P_o}\right)^{\frac{\gamma-1}{\gamma}}\right\}} + \left(\frac{P_x}{P_o}\right)^{\frac{\gamma-1}{\gamma}} \Big/ \sqrt{\frac{2\gamma}{\gamma-1}\left\{1-\left(\frac{P_x}{P_o}\right)^{\frac{\gamma-1}{\gamma}}\right\}}\right] \times \sqrt{\frac{G\theta}{M}} \quad .. \quad (4)$$

$= F_1 \times F_2$ for convenience.

It will be seen from the above that the exhaust velocity can be considered as the product of two separate functions. The first of these is dependent on γ and the expansion ratio P_o/P_x while the second is dependent only on $\frac{\theta}{M}$. It should be observed that in free space, v_e' is independent of the combustion pressure P_o and the exhaust pressure P_x considered separately, but depends only on the ratio of these quantities.

The F_1 and F_2 factors are shown plotted against $\frac{P_o}{P_x}$ and γ, and θ and M, respectively, in Figs. 1 and 2 (page 254). The beneficial effect on the exhaust velocity of increasing $\frac{P_o}{P_x}$ and θ and decreasing M is apparent from these curves. Shown chain-dotted on Fig. 1 is the corresponding curve of F_1 for $\gamma = 1{\cdot}2$, assuming that expansion takes place not in a vacuum but down to the ambient pressure in each case. The difference between the two $\gamma = 1{\cdot}2$ curves indicates that between 10 and 15 per cent. improvement in effective exhaust velocity and fuel consumption can be obtained when operating in vacuo, compared with operation at the same pressure ratio down to an ambient pressure equal to the exhaust pressure.

Also indicated (as a dashed line for $\gamma = 1{\cdot}2$ again) is the value of F_1 obtained by putting $\frac{P_o}{P_x} = \infty$. The value of v_e' corresponding to this value of F_1 is that which would be obtained using the simple equation $v_e' = \sqrt{2gHJ}$, where H is the calorific value of the propellant combination and J the mechanical equivalent of heat. The improvement in F_1 between this curve and that for practical expansion ratios (about 40 per cent. for $\frac{P_o}{P_x} = 20$ and as much as 15 per cent. even for $\frac{P_o}{P_x} = 1{,}000$) is a measure of the amount one can be misled when determining exhaust velocities by merely assuming that all the chemical energy can be converted into kinetic energy.

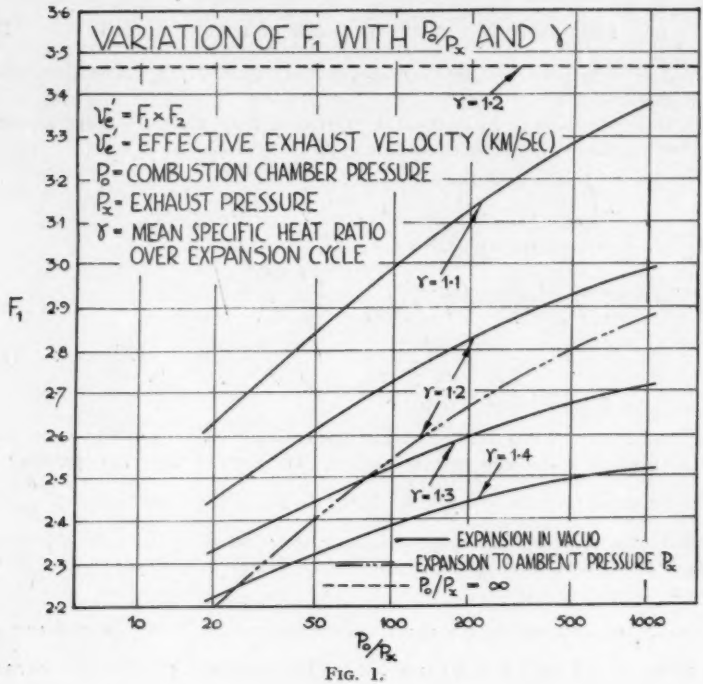
VARIATION OF F_1 WITH P_o/P_x AND γ
$V_e' = F_1 \times F_2$
V_e' - EFFECTIVE EXHAUST VELOCITY (KM/SEC)
P_o - COMBUSTION CHAMBER PRESSURE
P_x - EXHAUST PRESSURE
γ - MEAN SPECIFIC HEAT RATIO OVER EXPANSION CYCLE
γ = 1·2
γ = 1·1
γ = 1·2
γ = 1·4
γ = 1·3
EXPANSION IN VACUO
EXPANSION TO AMBIENT PRESSURE P_x
$P_o/P_x = \infty$
F_1
P_o/P_x
3·6
3·5
3·4
3·3
3·2
3·1
3·0
2·9
2·8
2·7
2·6
2·5
2·4
2·3
2·2
10
20
50
100
200
500
1000

FIG. 1.

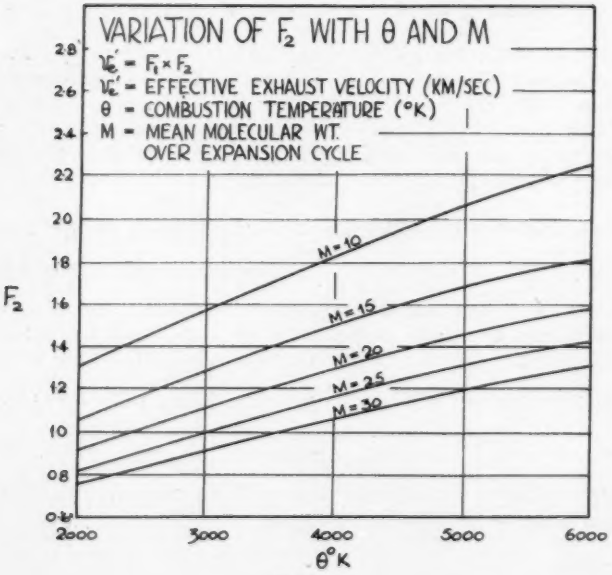
VARIATION OF F_2 WITH θ AND M
$V_e' = F_1 \times F_2$
V_e' = EFFECTIVE EXHAUST VELOCITY (KM/SEC)
θ = COMBUSTION TEMPERATURE (°K)
M = MEAN MOLECULAR WT. OVER EXPANSION CYCLE
M = 10
M = 15
M = 20
M = 25
M = 30
F_2
θ°K
2·8
2·6
2·4
2·2
2·0
1·8
1·6
1·4
1·2
1·0
0·8
0·6
2000
3000
4000
5000
6000

FIG. 2.

Such values of exhaust velocity cannot be realized in practice, because it can never be practicable to design for an infinite expansion ratio or in fact possible to convert from one form of energy to another at 100 per cent. efficiency.

Having established that the effective exhaust velocity is a function of γ, M, $\frac{P_e}{P_x}$ and θ, let us now consider these effects one by one and the limiting values of each which it appears we shall be able to use.

γ and M.

Both γ and M can be dealt with quite quickly. They are both physical properties of the gaseous products of combustion. Both will thus depend on the propellant combination used and can be further varied by adjusting the mixture ratio in which these are burnt. However, the effect on γ of doing this is so small, and will probably be so masked by other changes which will occur (such as one of temperature), that steps to vary γ intentionally need not be considered further.

Mean molecular weight, on the other hand, can have considerably more effect on v'_e because it can be intentionally varied more easily. Its value will vary for different propellant combinations, and those which give rise to products with low values of M show considerable advantage on this score. For example, water vapour formed when oxygen and hydrogen are burned stoichiometrically has an undissociated molecular weight of 18, whereas the mean molecular weight of the products of (say) nitric acid and kerosene (again burned stoichiometrically and again assuming no dissociation) is about 28. Therefore, everything else being equal, a 25 per cent. improvement in v_e' would be obtained by the use of the low molecular weight combination. The effect of this parameter can also be used to advantage in another way, since it is affected not only by the choice of propellant combination, but also, as stated earlier, by the ratio in which they are burned. If one of the propellants burned is appreciably lighter than the other, it may pay not to burn at the stoichiometric mixture ratio but to use the lighter component in excess, thereby decreasing the mean molecular weight of the combustion products. The reduction in molecular weight thus obtained may more than counterbalance the fall in combustion temperature and an improved performance will result. This is particularly the case with the hydrogen/oxygen and hydrogen/fluorine reactions, and curves showing v_e' values, as a function of combustion temperature while the mixture ratio is altered, are presented in Fig. 3. The fact that the highest values of v_e' do not occur at the highest temperatures is almost entirely the result of molecular weight changes. This effect is of course especially important with regard to the cooling of the motor, since it indicates a way in which the highest exhaust velocities can be obtained without having to operate at the maximum temperatures associated with the use of stoichiometric mixture ratios. It must be made clear, however, that such a reduction in operating temperature is not obtained without an attendant disadvantage. This arises from the fact that as the proportion of the lighter propellant is increased, the mean density of the unburnt propellants decreases, aggravating the problem of designing light tanks to accommodate them. This is shown up by the dashed curves, which show

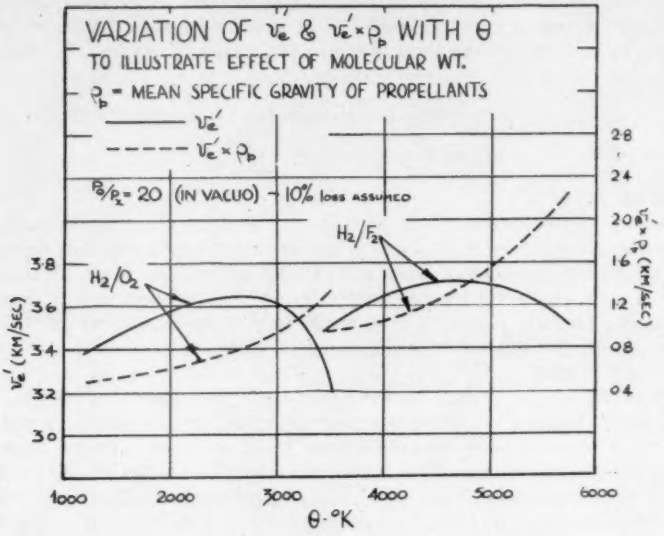
VARIATION OF v_e' & $v_e' \times \rho_p$ WITH θ
TO ILLUSTRATE EFFECT OF MOLECULAR WT.
ρ_p = MEAN SPECIFIC GRAVITY OF PROPELLANTS
v_e'
$v_e' \times \rho_p$
P_0/P_x = 20 (IN VACUO) – 10% loss assumed
H_2/F_2
H_2/O_2
38
36
34
32
30
28
24
20
16
12
08
04
1000
2000
3000
4000
5000
6000
θ·°K
v_e' (KM/SEC.)
$v_e' \times \rho_p$ (KM/SEC)

FIG. 3.

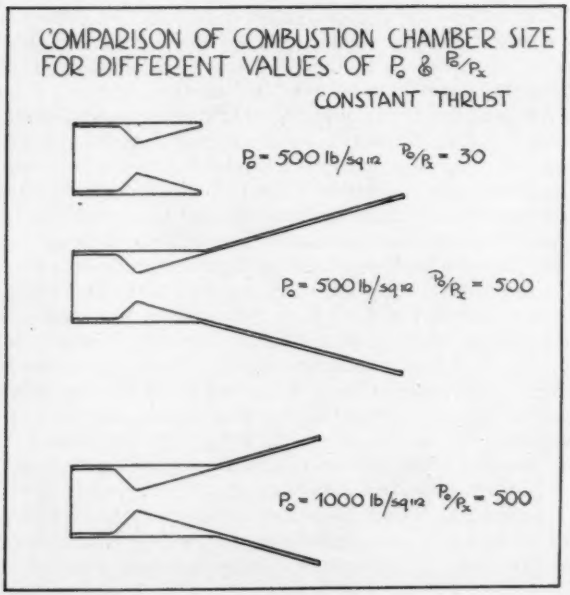
COMPARISON OF COMBUSTION CHAMBER SIZE
FOR DIFFERENT VALUES OF P_0 & P_0/P_x
CONSTANT THRUST
P_0 = 500 lb/sq in P_0/P_x = 30
P_0 = 500 lb/sq in P_0/P_x = 500
P_0 = 1000 lb/sq in P_0/P_x = 500

FIG. 4.

how $v_e' \times$ (mean specific gravity of propellants) varies, this factor being proportional to the volume of propellants required for a particular mission, and thus indicating the size of tanks which will be needed.

Expansion Ratio and Pressure

Fig. 1 has indicated the way in which the exhaust velocity obtained depends on the pressure ratio over which the combustion gases are expanded. The greater this expansion ratio is, the lower the exhaust temperature of the gases will be and the greater will be the amount of thermal energy which is converted to useful kinetic energy.

It has been shown (Eqn. 4) that for operation *in vacuo*, the exhaust velocity is independent of the absolute value of the chamber pressure (apart, of course, from the secondary effect that this has on the amount of molecular dissociation which occurs and the effect which this then has on the temperature, and consequently on the exhaust velocity). Although the combustion pressure does not directly affect the exhaust velocity it must not be assumed that its choice becomes a completely free one, because it can be shown that under certain conditions, a high expansion ratio can be profitably obtained only by the associated use of a high combustion pressure.

This arises from the fact that the employment of a high expansion ratio with a low chamber pressure results in a chamber and nozzle of very large dimensions. On the other hand, an increase in chamber pressure leads to a proportional decrease in cross-sectional area of both the chamber and nozzle, and also (if the nozzle divergence angle is maintained constant) to a decrease in nozzle length.

Fig. 4 indicates, roughly, to scale, how rocket motor sizes would vary for different chamber pressures and expansion ratios, for motors giving the same thrust. The first configuration represents what one might term good present day practice, i.e., a chamber pressure of 500 p.s.i. and an expansion ratio of 30. The second indicates how the size would change if the same pressure were used, but if the expansion ratio were increased to 500. The third sketch then illustrates how the size would further vary if the expansion ratio was kept at 500 but if the chamber pressure were increased to 1,000 p.s.i. These comparisons are only intended to serve as a rough guide to indicate how much more space would be occupied by the motor if high chamber pressures are not used to obtain high expansion ratios. They give no indication, of course, of how the relative weights will vary for the different conditions. Although the higher pressure motor appears smaller in outline, it will have to be fabricated in a more robust manner to withstand the greater loads, and a complete weight analysis for the different cases proves to be both a complex and conjectural undertaking.

Such a study becomes rather speculative because it is difficult to decide on exactly what assumptions one should make comparisons, since the nature of these will depend to a great extent on the way in which rocket motor design evolves.

Very broadly, it is permissible to divide the complete motor into two main groups, the pumps, turbines, pipes and valves falling in one, and the combustion chamber in the other. The weight of the first group will fairly clearly increase

in some way with pressure. A study of existing systems together with an idealized design investigation would indicate that the increase would not be quite proportional to pressure, but would vary as some lesser power of it. For the purpose of this paper it has been assumed that the weight of all the parts of the motor other than the combustion chamber will vary approximately as $P_o^{\frac{1}{2}}$. The comparison has been made, of course, on a basis of constant thrust, but the improvement in exhaust velocity at higher expansion ratios and the resultant secondary decrease in propellant flow rate has been considered sufficiently small to permit its effect on pump weight to be ignored.

With regard to the chamber weight, the overall proportions of this component are easily determined for each design. The corresponding variation in wall thickness, however, is harder to assess, and will depend on the way in which the chamber is constructed. In this assessment, it has been assumed that the chamber inner wall would not get thicker as pressures and diameters increase but would remain constant regardless of the size of the chamber and the pressure at which it is operating. This is considered to be a specific requirement for effective cooling (see later section) and could be achieved by such methods as balancing out the loads on the wall by equalizing the pressures of the gas inside and the coolant outside, or preferably, by relieving its stresses by supporting it by some means from the cool outer wall. It has also been assumed that regenerative cooling is used and that the outer wall has to withstand the combustion pressure plus a proportional injector and feed line pressure loss. It has been taken that stress levels for corresponding components are constant throughout, while no particular allowance has been made for the proportional reduction in scantling size for the bigger motors. The comparison has, for convenience, been based on a 60,000 lb. thrust motor, it being argued that a higher total thrust could probably best be obtained by a multiplicity of such motors. In any case, the actual size of the motor considered would not greatly affect the conclusions reached.

Fig. 5 records the results of this study showing how the complete rocket motor weight might vary with chamber pressure and expansion ratio. The curves have been plotted on a comparative non-dimensional basis, all the weights being ratioed to that of a motor with a chamber pressure of 500 p.s.i. and an expansion ratio of 30.

The general inference to be drawn is that for expansion ratios less than 100, very little appears to be gained, as far as weight is concerned, by employing chamber pressures greater than those in use to-day. For ratios of the order of 500–1,000, however, there is a considerable advantage in increasing the chamber pressure up to values far above those to which we are at present accustomed. The weight penalty incurred by the use of such high ratios is, however, considerable, regardless of the pressures used, and if pressures as low as 500 p.s.i. are employed it will be so great as almost certainly to prohibit their use.

Whether the weight increases incurred are worthwhile or not can be decided only by a detailed study of their effect on the performance of particular projects. In such a study exhaust velocity changes, resulting from variations in expansion ratio must also be taken into account.

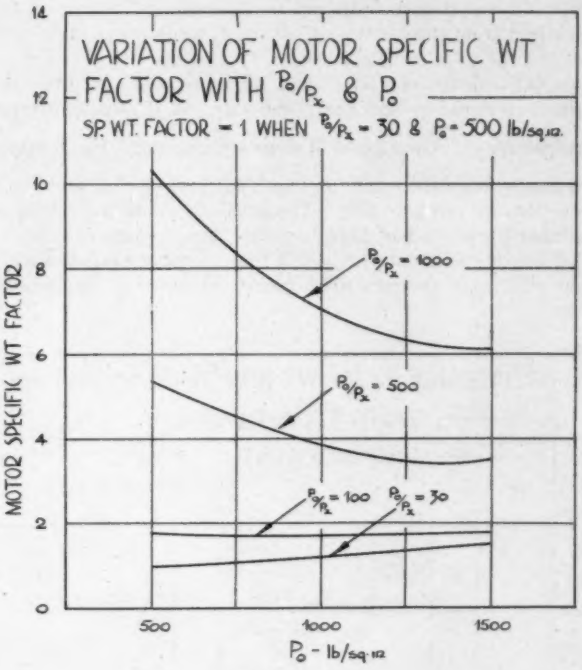

FIG. 5.

Two of the aforementioned specific cases were therefore considered in this light:—(i) the orbital case and (ii) the orbit-Moon-orbit expedition. In each case the vehicle structure and tank weight was assumed to be constant and equal to 16 per cent. of the propellant weight (cf. 19 per cent. for V2), while the rocket motor was assumed for convenience to have the following characteristics:

$$\theta = 5{,}000^\circ \text{ K.}$$
$$M = 20$$
$$\gamma = 1{\cdot}2$$
$$P_o = 1{,}000 \text{ p.s.i.}$$

No combustion or nozzle friction losses were included, while the specific weight of the "basic" motor of Fig. 5 (i.e., one of 500 lb./sq. in. and expansion ratio = 30) was assumed to be ·025 lb. wt. per lb. of thrust (representing an anticipated improvement over the corresponding approximate figure of ·036 for the V2 motor).

In Case (i), three steps and an initial acceleration of $1g = 32$ ft./sec. were assumed. The effect of varying expansion ratio on the initial weight required to carry one ton of payload is shown in Fig. 6, which indicates than an expansion

ratio of only 55 would give optimum performance. The use of a chamber pressure different from the 1,000 p.s.i. assumed, would have little effect on this result.

For Case (ii), calculations have been made for two variants, one with 16 ft./sec. initial acceleration and the other with 3·2 ft./sec. (corresponding to $\frac{g}{2}$ and $\frac{g}{10}$ respectively). Once again, 3 steps are assumed. Fig. 6 shows that for the greater acceleration the optimum expansion ratio is 275, while for the lower acceleration, this increases to 600. These values would have been greater, if higher chamber pressures had been assumed, but because of other considerations, which are discussed later, it is felt that chamber pressures when used in conjunction with high temperatures, might, in practice, be limited to even lower values.

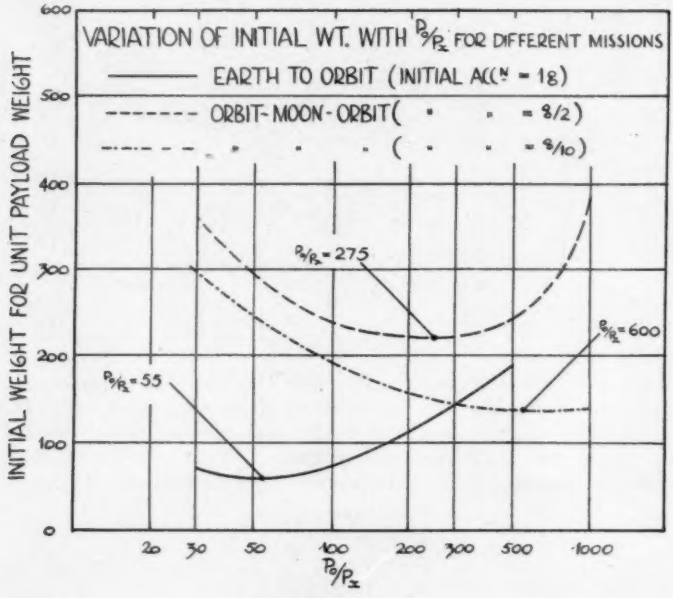

FIG. 6.

These curves are interesting since they show that, according to the assumptions on which they are based, the optimum expansion ratio for the orbital rocket is not far removed from present day practice. This is particularly fortunate, since there appears to be a definite upper limit to expansion ratio for the motor of a vehicle intended for this duty, dictated by its cross-sectional area. Assuming, for example, that it was designed, because of aerodynamic considerations, to a "fineness ratio" similar to that of V2, it can be shown that

the limit of expansion ratio it is possible to use is dependent on the size of the ship, and will probably get smaller as the weight of the ship increases. If this was assumed to weigh 500 tons at take-off, and have an overall density of 30 lb./cu. ft. (about the same as V2 had), then the limit would be 200 to 1 for a chamber pressure of 1,000 lb./sq. in. If the chamber pressure were reduced to 500 lb./sq. in., then the expansion ratio would be correspondingly further restricted to about 75 to 1.

For vehicles of lower accelerations, the optimum expansion ratios are greater, since the weight of the propulsive unit becomes less compared with the total weight and a greater per cent. weight penalty can be tolerated for a given per cent. performance increase. This effect is, of course, further demonstrated by the even greater optimum expansion ratio which is obtained if the acceleration is limited to the lower value.

The actual values for optimum expansion ratios obtained will, of course, depend not only on the rocket thermodynamic conditions assumed, but also on the assumptions made to determine how the motor weights vary for different conditions. The results which have been presented, therefore, only illustrate a general trend which must exist and show that for expeditions not involving an ascent from the earth's surface, expansion ratios considerably greater than those currently used, may become accepted practice, if chemical rockets are to be used for this purpose.

Any considerable increase in expansion ratio or chamber pressure will inevitably create new problems or tend to aggravate those that already exist.

Cooling

For example, a higher expansion ratio in requiring a larger nozzle, will increase the area which has to be kept cool. The problem of keeping the wall of the nozzle at an acceptable temperature level may not itself be especially severe for the following reasons. The radiated heat transfer per unit area of the nozzle will be low, because of the comparatively low bulk temperature of the highly expanded gases, while the convected heat transfer rate will be correspondingly restricted by the thick protective boundary layer which will tend to build up in a long nozzle. On the other hand, the actual amount of heat which is transported *to*, and has to be carried away *by*, the coolant will increase, simply because of the greater area through which the heat can flow. The problem of having to cool with a propellant, which is probably an unsuitable coolant anyway, will thus be made more difficult.

The use of high expansion ratios will also increase the friction loss in the nozzle and mean that a greater amount of kinetic energy will be converted back to heat energy and will be lost as far as the attainment of useful exhaust velocity is concerned. While present-day values of nozzle efficiency of about 90 per cent. (based on energy) are achieved, it is possible that lower values than this will have to be accepted if higher expansion ratios are used. A decrease in nozzle efficiency with expansion ratio would, of course, influence the optimum values calculated for different missions, and would reduce them slightly in each case.

An increase in operating pressure as opposed to expansion ratio would have both its advantages and disadvantages. Firstly, most chemical reactions tend to take place more rapidly at high pressures. This might permit the use of smaller values of L* (resulting in smaller combustion chambers), which in turn would reduce the quantity of heat transferred to the coolant. At very high chamber temperatures, when a much larger part of the heat flow is by radiation and occurs in the chamber, this might be an important consideration. In addition, shorter reaction times would result in more complete and steadier combustion, leading to higher combustion efficiencies. Whereas these at present range between 90 and 95 per cent., higher values of the order of 97–98 per cent. might be expected when increased combustion pressures are used. This especially applies to inherently energetic propellant combinations such as hydrogen and oxygen or hydrogen and fluorine.

With regard to the question of cooling the combustion chamber, any increase in operating pressure will result in an increase in heat transfer rate, which in turn would lead to a rise in wall temperature. Some estimated figures for a hypothetical chamber designed to operate at a temperature of about 4,000° C. illustrate this effect. An increase in operating pressure from 500 to 1,000 p.s.i. would result in an increase in average heat flow rate (per unit area of the chamber) of about 50 per cent. and an increase in wall temperature at the throat of about 200–300° C. However, for a given expansion ratio, the total area to be cooled would go down by about 45 per cent., so that the total amount of heat transferred to the coolant might actually be less at the higher pressure. Over and above this, if regenerative cooling were used, the coolant would have a higher boiling point, and so would be able to absorb the heat more satisfactorily. Thus, it would seem from this cursory study, that using a higher operating pressure would ease the cooling problem as far as the coolant is concerned, but would at the same time result in higher wall temperatures. It would therefore seem possible that this latter consideration might well limit the chamber pressure (and thus indirectly the expansion ratio) which could be used, and that this limitation might be more severe with high temperature propellant combinations.

To sum up, therefore, it appears that the optimum expansion ratios differ for different missions, being not far removed from present day practice when the weight of the propulsive unit is a large percentage of the weight of the vehicle it propels. It follows that for this case, modest values of chamber pressure are also likely to be used. When the propulsive unit is comparatively smaller, however, optimum values of expansion ratio will be greater, approaching a figure perhaps 10–20 times current practice. Higher expansion ratios will probably entail a cooling problem of greater severity due to the greater area to be cooled, and for the same reason increased nozzle losses due to friction will have to be allowed for. These high expansion ratios are likely to be obtained in conjunction with the use of higher chamber pressures, approaching 1,000 p.s.i. or else combustion chamber weights become excessive. If pressures of this order are used, the problem of keeping the chamber cool, is further aggravated, but higher combustion efficiencies might be expected which will tend to counteract the increased nozzle losses.

Temperature

The temperature at which the propellants are burnt can probably have a greater influence on the exhaust velocity than any other single factor. On the other hand, any substantial increase in operating temperature will aggravate the already severe fundamental problem of cooling. The ultimate solution of this problem may depend on completely new techniques and the employment of heat resisting materials far better than those available at present.

The problem of heat transfer is two-fold. Firstly, the temperature of the chamber wall must not at any point exceed a value at which it is likely to melt, burn away, or soften sufficiently to collapse under load. Secondly, the heat passing through the chamber wall must be safely carried away by some cooling medium which will satisfactorily absorb it and not undergo any undesirable physical or chemical change in the process.

At the present time, chamber temperatures have been limited to much less than 3,000° C. for rocket motors which have had to operate repeatedly or continuously. This limitation has been entirely a structural one, dictated by the heat resisting properties of the materials from which the chambers have been made. It has rarely been imposed by the limiting thermo-chemical properties of the particular propellant combinations used, since these have usually been burnt at off-stoichiometric mixture ratios, merely to limit the temperatures to values which could be tolerated.

To operate at this order of temperature, combustion chambers without exception have been regeneratively cooled, passing either the oxidant or the fuel round the chamber before it is injected and burnt. The heat transfer rates (of the order of 200–300 CHU/sq. ft./sec. averaged over the whole chamber and 750–1,000 CHU/sq. ft./sec. near the throat where the rate is highest) have been low enough to permit the use of one of the propellants for this purpose, without undue risk that it would boil or decompose. By this means the temperature of the chamber wall in contact with the hot combustion gases has been kept down to about one-third of the gas temperature, even at the throat where the heating condition is most severe.

Heat is passed from the combustion gases to the chamber wall by two distinct processes, by convection within the gaseous boundary layer adjacent to the wall, and by radiation. As the combustion temperature rises the amount of heat conveyed by each process will rapidly increase, and this will, of course, result in a corresponding increase in wall temperature. Because of this it does not appear feasible that combustion temperatures greater than about 3,500° C. can be employed, using simple regenerative cooling means as described above. This limitation has been based on the assumption that maximum wall temperatures must be kept below about 1,500° C., at which temperature most of the most suitable materials in general use would begin to melt or at least soften to such an extent that they would be quickly eroded away by the scrubbing action of the expanding gases.

In order to accommodate the higher temperatures resulting from the use of more energetic propellant combinations, there are a number of detail design

improvements which can be made, each of which could affect to advantage the heat transfer process through the chamber wall.

For example, the velocity of the coolant circulating through the jacket surrounding the chamber can be increased. This will have the effect of decreasing the thickness of the boundary layer on the coolant side, and of bringing the wall temperature nearer to that of the relatively cold coolant. Unfortunately, although it goes in the right direction, the resultant change in wall temperature is small (amounting to about 20° C. on the hot side of the wall for a representative case, assuming that the coolant velocity is doubled) and is furthermore accompanied by a small increase in the actual amount of heat transferred. Additionally, of course, any increase in coolant velocity will increase the pressure drop through the coolant jacket, and a larger and heavier pumping system will be required to overcome this.

A slightly more effective way of reducing the wall temperature, again with its peculiar disadvantages, is to reduce the thickness of the chamber wall itself. For the particular example considered, reducing the wall thickness from 0·1 in. to 0·05 in. would reduce the maximum wall temperature by about 350° C., but at the same time the heat flow through the wall would be increased by nearly 30 per cent. Even if this increased heat flow could be satisfactorily absorbed by the coolant, such a thin chamber wall would not have sufficient strength, especially at its high temperature, to withstand any appreciable differential pressure loads. Steps would have to be taken, therefore, to relieve the chamber of its stresses, either by carefully regulating the coolant pressure with respect to the gas pressure inside the chamber, in order to balance out the load on the wall at its most critical regions, or else by providing support for it, by connecting it to the cool outer wall of the coolant jacket.

A much more effective method of easing the temperature problem, in all respects, is to increase artificially the thickness of the gaseous layer adjacent to the chamber wall, since this would have the effect of reducing at the same time both the wall temperature and the amount of heat transferred. The boundary layer thickness can be increased to give this result by continuously introducing through the chamber wall either a cool gas, or liquid which will then evaporate and augment the natural boundary layer. Film or sweat cooling are variants of this technique which have been successfully employed. For the representative example considered, doubling the thickness of the boundary layer would have the effect of decreasing the hot wall temperature by about 600° C. and reducing the heat transfer rate by nearly 50 per cent. Film cooling was, of course, effectively used in the V2 combustion chamber, in which alcohol was bled into the chamber through a number of small holes adjacent to the throat. A more effective method of achieving a similar result would have been to percolate coolant through a porous chamber wall, instead of using a number of discrete holes. "Sweat" cooling in this way, would ensure more even distribution of the coolant, which additionally would cool the chamber wall throughout its thickness.

Up to now, the development for such a process of porous metals which would have sufficient strength and could be made in the necessary shapes and sizes, has not yet advanced sufficiently to enable it to be carried out on any large scale.

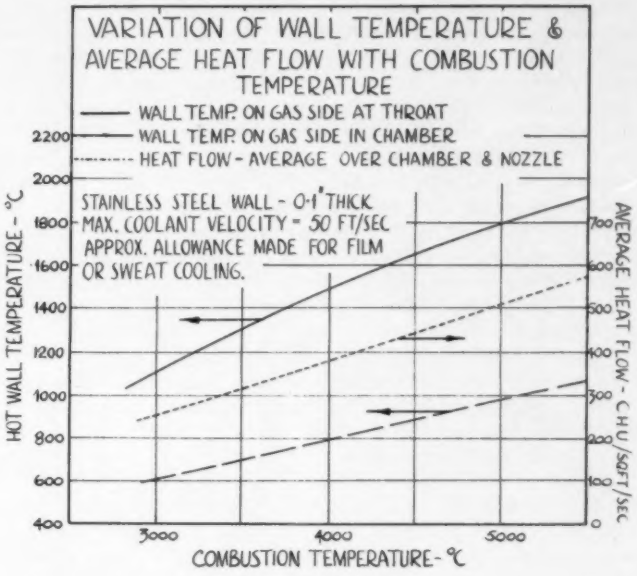

FIG. 7.

Laboratory experiments, however, have already demonstrated the effectiveness of this technique, which has the added advantage that when the liquid propellant which is injected is evaporated adjacent to the chamber wall, the absorption of latent heat will further decrease the temperature of the boundary layer and therefore that of the wall itself.

Assuming either film or sweat cooling is employed, the variation of wall temperatures in the chamber and at the throat have been calculated for a range of combustion temperatures, and are shown in Fig. 7. No allowance has been made for latent heat absorption in the gaseous boundary layer, or nuclear boiling of the coolant, and on both of these counts the values obtained will be conservative. Added to this, it has had to be assumed that low temperature heat transfer laws apply throughout, although there is no experimental evidence as yet that this will be the case, and wildly extrapolated figures for viscosity, thermal conductivity, etc., have had to be assumed. The results shown therefore are only indicative of a trend and may be to some extent in error, especially at the higher temperatures. With this uncertainty in mind, therefore, it would appear that combustion temperatures, using existing materials together with provision for sweat cooling, will be limited to about 4,000° C., the criterion still being the maximum wall temperature at the throat. It is interesting to note, in this respect, that even at very high combustion temperatures, the throat condition is still the most critical, in spite of the greatly increased radiated heat flow in the chamber.

In order that still higher combustion temperatures can be used, and greater values of exhaust velocities obtained, it would appear that the development of better heat resisting materials with improved strength and scaling properties will be necessary, and that these will be used in conjunction with the detail design refinements to which reference has been made. In this category of improved materials, those which promise the greatest ultimate improvement appear to be of the refractory type. Up to now these have been used only for certain specialized applications, because their poor thermal conductivity and brittleness have resulted in inadequate thermal shock properties. The development of metal/ceramic mixtures (ceramals or ceramets) promises an improvement in these essential properties. If one can assume the eventual use of a low stressed, porous, sweat cooled, ceramet wall which could operate at face temperatures approaching 2,000° C., then it would appear possible to use combustion temperatures of the order of 5,000° C.

Up to now, the effect of higher temperatures on the coolant itself has been largely ignored. As higher temperatures are used, more heat will be transferred to the coolant, until eventually the temperature rise through the jacket will be so great that general boiling will occur, with a breakdown in the heat transfer process. The increase in average heat flow density (in CHU/sq. ft./sec.) with combustion temperature is shown in Fig. 7, from which it will be seen that as this increases from 3,000° C. to 5,000° C., the heat transfer rate for the same basic size of motor will approximately double in amount. An attendant increase in operating pressure (by weakening the extent of the gaseous boundary layer) and an increase in expansion ratio (by increasing the area to be cooled) would further increase the amount of heat transferred.

With comparatively inefficient coolants (e.g., liquid oxygen or hydrogen), it is difficult to imagine how such a quantity of heat could be absorbed satisfactorily. In such cases, it might be necessary to use a secondary coolant fluid, water being an obvious example which might be considered. The heat from this secondary coolant might, for one such possible system, be removed in the form of mechanical work, utilized for auxiliary power or perhaps to drive the propellant pumps, as was proposed in Ref. 7.

To recapitulate, very briefly, then, it seems probable that with the use of existing techniques and materials, combustion temperatures may well be limited to about 3,500° C. Assuming, however, the development of unstressed chamber walls made of improved heat resistant alloys, or refractories which will resist thermal shock, then temperatures of the order of 5,000° C. may be ultimately used. Because of the greater amount of heat which will be radiated to the chamber walls, especially upstream of the throat section, film or sweat cooling will almost certainly have to be applied to the whole area to be cooled.

The use of higher chamber pressures and expansion ratios will further aggravate the cooling problem, mainly because the former will increase the heat flux and hence the wall temperature, while the latter will increase the area to be cooled. Since the high temperatures necessary for the attainment of high exhaust velocities will most likely be obtained with low-boiling-point, low-specific-heat propellants (which are also probably highly corrosive), some special cooling fluid may have to be introduced.

Miscellaneous Factors

There are a number of additional factors that affect the choice and determine the overall effectiveness of a propellant combination, other than just the attainment of a high exhaust velocity. Perhaps the most important of these is the effect of the mean propellant density on the tank weight. This consideration has particular bearing on the ultimate usefulness of the chemical rocket, since most of the chemicals which give the best thermodynamic performance, tend to fall into the low density range.

The extent to which low density propellants are a disadvantage depends on the critical factors which dictate the tank design. To illustrate this, a factor w_t has been introduced, this being the tank specific weight, i.e., the weight of the tank divided by the propellant weight it carries (W_p). Assuming the tanks are of the same geometrical shape, it follows that if they are designed to withstand a constant internal pressure, $w_t \propto \frac{1}{\rho}$ where ρ is the mean propellant density. This relationship would probably apply if the inertia loads on the tank contents were low, as they would be in acceleration away from an orbital refuelling base. A small constant internal pressure is then not an unrealistic design assumption, since this would be necessary to suppress boiling and to keep the pumps satisfactorily primed. For such a duty a spherical tank would represent the optimum shape. If on the other hand the tanks were designed for a given large acceleration, as for example during the take-off from the Earth's surface, the inertia forces would generate greater stresses in the tank wall than those imposed by a small constant internal pressure. Under these circumstances, it can be shown that the pressure generated by the head of liquid will decide the wall thickness. It can then be shown that, assuming a given shape of tank, wall material and stress:—

$$W_p \propto V_p \rho \propto d^3 \rho$$

(where V_p is the volume and d a linear dimension of the tank). The pressure at any point in the tank p is then $\propto d\rho$ and the wall thickness $\propto pd$ and hence $\propto d^2\rho$.
The weight of the tank is $\propto d^2 \times d^2\rho = d^4\rho$.
The specific wt. w_t is then $\propto d^4\rho/W_p \propto d^4\rho/d^3\rho \propto d$
But $d \propto (W_p/\rho)^{\frac{1}{3}}$ hence $w_t \propto (W_p/\rho)^{\frac{1}{3}}$.

Whichever assumption is accepted, a low propellant density always results in a heavier tank, which in turn would lead to a secondary increase in structure weight. In addition to this, a further increase would occur for the Earth take-off case, at any rate, if low boiling point propellants were used, due to the lagging required. For operation in space, however, this might not be such a serious problem, since external heat could only be *radiated* into the tanks, and its amount might be restricted by suitably treating their external surfaces. Means would also have to be provided, however, to prevent further heat being conducted to the propellant from the motor compartment, etc.

No specific mention has yet been made to any possible limit to the size of chemical rocket motors, and the remarks which have been made up to now have

not applied to any one particular size more than another. It is quite possible, however, that although there appears to be no very hard and fast reason why size should be limited, very large thrusts will be obtained not by a single large motor but by a combination of small ones. The reasons suggested for this are as follows:—

(1) Small motors will be easier and cheaper to make and develop and will require less elaborate and expensive test equipment.
(2) Different missions could be accomplished with different combinations of such small units, so that a different propulsive unit will not have to be developed for each application.
(3) A multiplicity of such units will give a greater operating reliability than a single unit.
(4) For very large motors it might become more difficult to attain such a low specific weight as for smaller motors.
(5) The cooling problem might become more difficult for large motors, due to the reduction in boundary layer width at high Reynolds Nos. This trend particularly affects conditions in the expansion nozzle and will therefore apply with greater force when high expansion ratios are used.

A brief mention must also be made of the means by which the propellant pumps are driven. It is essential that this system is as efficient as possible, since the per cent. flow of propellant required to drive the pumps, unless this is put to some useful propulsive effect, is equivalent to the same per cent. loss in exhaust velocity. For chamber pressures of the order of 300 to 500 p.s.i., about 1 to 5 per cent. of the propellant goes to drive the pumps and since this flow is roughly proportional to chamber pressure, the need for economy is particularly vital if higher pressures still are to be used. The problem becomes even more critical when low density propellants are used, since the pumping power is proportional to the propellant volume rather than the propellant weight. For the missions considered a 5 per cent. loss in exhaust velocity (e.g., resulting from its equivalent in pump consumption) would result in anything up to 50 per cent. increase in initial weight for the same payload. It would therefore seem very necessary that a system be used which at least is as efficient as possible and also in which the turbine throughput is not thereafter wasted, but is put to some useful propulsive effect. Such a means was referred to in Ref. 8.

Characteristics of Known Propellants and their Applications to Specific Projects

Up to now the increases in exhaust velocity which might be obtained with the chemical rocket have been discussed in the light of some of the physical difficulties which will have to be solved if they are to be achieved. Only a passing reference has been made as yet to the propellant combinations it will be necessary to use in order to make these improvements possible. A complete list of all the possible reactions which occur in nature and which might be considered for use in a rocket motor would indeed be a formidable proposition, since any energy-carrying substance, used in conjunction with another substance

TABLE I

PERFORMANCE OF DIFFERENT PROPELLANT COMBINATIONS

	Propellants	M.R.*	Mean sp. gravity	v_e'†	v_e'‡	Comb. temp.	
A	Hydrogen peroxide—hydrazine	1·4 : 1	1·24	2·75	2·85	° C. 2,400	A
B	Oxygen—hydrogen ..	8 : 1	0·42	3·65	3·8	3,200	B
C	,, ..	4 : 1	0·28	4·15	4·25	2,700	C
D	Fluorine—hydrogen	18·9 : 1	0·64	4·0	4·15	5,400	D
E	,,	9·4 : 1	0·37	4·25	4·35	4,400	E
F	,,	7 : 1	0·29	4·15	4·25	3,700	F
—	Oxygen—lithium ..	1·2 : 1	0·66	3·65	3·75	7,000	—
—	Atomic hydrogen ..	—	—	11·0	11·2	4,700	—
G	"X" + "Y" ..	—	1·5	5·1	5·25	4,700	G

* Mixture ratio—Oxidant to fuel by weight.
† In km./sec. for $Po/Px = 50$ in vacuo. 10 per cent. energy loss.
‡ In km./sec. for $Po/Px = 250$ in vacuo. 20 per cent. energy loss, because of greater friction loss in longer nozzle.

with which it will combine to release all or part of its energy, can be considered as a legitimate possibility.

If such a list were prepared it could be divided broadly into three parts, and perhaps fortunately for anyone who might be persuaded to undertake the task, by far the greater part would consist of reactions which are much too ineffectual to be worth considering from the point of view of rocket propulsion, since either the energy released is too small, or the products of combustion too heavy.

The second of the sub-divisions is shorter and of more interest since it contains a number of propellant combinations which either are already being exploited in the missile or aircraft fields, or could be with only a little further development of existing techniques. It is unfortunate that almost all these reactions appear to just fail to make the grade as far as promoting interplanetary flight is concerned. Predominant among them are liquid oxygen, hydrogen peroxide and nitric acid as oxidants and various hydrocarbons, amines, and alcohols as fuels. Sea level exhaust velocities at the expansion ratios in general use, range between 2 and $2\frac{1}{2}$ km./sec., combustion temperatures attain values between 1,700° C. and 3,200° C., while the mean bulk specific gravity is usually about that of water. The best fuel in this group appears to be hydrazine, which gives about the same performance when used with any one of the three usual oxidants. The performance of the hydrazine/hydrogen peroxide reaction is given in Table I as being, as it were, about the most attractive that the present generation of propellants can offer.

The third and by far the shortest sub-division includes a small number of reactions, which, since they give appreciably higher exhaust velocities than those of the second group, are especially interesting as possibilities for interplanetary flight. Their superior performance is, however, generally given at the expense of higher operating temperatures and lower bulk densities. The two oxidants most concerned are liquid oxygen and liquid fluorine, while the most attractive fuel is liquid hydrogen, with some of the lighter metals and metallic hydrides, as second favourites. Liquid ozone might be considered as a slightly more effective but more expensive alternative propellant to liquid oxygen.

The exhaust velocities given by some of these reactions are given in Table I together with some physical data relating to them. Comparative data for some of these, and numerous other propellant combinations are given in Refs. 9, 10, 11 and 12. The velocities given here are for different expansion ratios and are effective (i.e., they assume that the operation takes place with no ambient pressure) while reasonable allowances have been made for combustion, friction and pumping losses. A greater friction loss has been assumed for the higher expansion ratios. The different values given for the oxygen/hydrogen and fluorine/hydrogen relate to different mixture ratios, the first two in each case corresponding to stoichiometric and maximum v_e' conditions respectively.

This table is only approximate, since to prepare it considerable extrapolation, interpolation and a lot of guesswork was used. Even so, it does indicate very clearly how much more effective are the reactions involving hydrogen as compared with those in present day use. It also shows the penalties which have to be paid, e.g., the much higher temperatures of the fluorine reactions and the poor bulk specific gravities when oxygen is used.

It will be apparent from the figures that the temperatures obtained with hydrogen and oxygen are not greatly above those which have already been used, especially when an excess of hydrogen is used. There would seem, therefore, to be no fundamental difficulty in the use of these propellants together, except that arising from the fact that neither of them is very suitable as a coolant. However, if an auxiliary coolant is used, that disadvantage would be obviated. To what extent the lower bulk densities associated with the lower-temperature mixture ratios are detrimental can only be decided by a more detailed study referred to a specific requirement. The use of fluorine, on the other hand, even at its optimum condition, imposes a severe temperature problem. Because of this, the necessity to run even richer in hydrogen must be considered, and the performance when the temperature is limited to 4,000° K. is included in Table I. Also included is an estimate of the lithium/oxygen reaction, as being an example of the use of a metallic fuel. Although these give rise to higher bulk densities than liquefied gases, it appears doubtful that they will find extensive use, because of the high molecular weight of their combustion products. Consequently high exhaust velocities can be obtained only at the expense of high temperatures.

The atomic hydrogen reaction is also included mainly for the sake of completeness. It must be made quite clear that its inclusion is an academic one, since as far as can be foreseen at present this reaction cannot be considered for

a rocket motor, because of the storage difficulty. Mon-atomic hydrogen cannot be contained in liquid form, and even in its gaseous form has an approximate life of less than a second before it reverts to its normal form. Apart from this extremely fundamental drawback, the temperature when it does react is quite high, especially compared with, say, the reaction of hydrogen and oxygen.

It is fairly unlikely that there are any natural exothermic chemical reactions as yet completely unknown, which would be suitable for use in a rocket motor. The information given in Table I therefore indicates the extent to which the performance of the chemical rocket motor is limited by the chemicals at our disposal, regardless of whether we shall be able to overcome the physical problems incurred when using them. Although we cannot envisage at the present time how synthetic fuels of a greater energy content than those in existence could be manufactured, the ultimate possibility of doing this is not without interest. If by inventing, so to speak, new molecules, it might also be possible to choose them, so that when they are burnt with suitable oxidants, which would also be "invented" for the purpose, then products of low molecular weight would result.

It would therefore seem reasonable to specify that such "tailor-made" propellants should have the following properties:—

(*a*) A combustion temperature which it might be possible to utilize without undue difficulty.

(*b*) Products of combustion with a low molecular weight (such as might be given by a large proportion of water vapour, together with some free hydrogen).

(*c*) A mean propellant density appreciably higher than that for known high performance propellants.

We might then expect that burning such a hypothetical oxidant "*X*" with an equally hypothetical fuel "*Y*" would give the following performance:—

$$\begin{aligned}
\theta &= 5{,}000^\circ \text{ K.} \\
M &= 10 \\
\gamma &= 1{\cdot}2 \\
v_e'(P_o/P_x = 50) &= 5{\cdot}1 \text{ km./sec.} \\
v_e'(P_o/P_x = 250) &= 5{\cdot}25 \text{ km./sec.} \\
\rho &= 1{\cdot}5
\end{aligned}$$

We shall proceed in a moment to see how such a combination would increase the field of usefulness of the chemical rocket. As stated at the beginning of this paper, the overall effectiveness of any particular propellant combination can be fairly assessed only by the study of its application to a particular project. For this reason the use of some of the combinations given in Table I have been assumed for the three missions which were postulated earlier. The results of this assessment are given in Table II, the weights given being initial values which would be required to transport one ton of payload.

Since we are now considering actual propellants, more realistic estimates can be made of the tank weights and the corresponding structure weights which might be expected. For convenience it has been assumed that the tank weights

TABLE II

INITIAL WEIGHTS TO CARRY UNIT WEIGHT OF PAYLOAD

	Propellants	Orbital case v_e = 10 km./sec. 3 steps		Orbit-Moon-orbit v_e = 16 km./sec. 5 steps		Orbit-Mars-orbit v_e = 30 km./sec. 5 steps							
		ϵ*	Wt.*	ϵ†	Wt.†	ϵ†	Wt.†	ϵ‡	Wt.‡	ϵ†	Wt.†	ϵ‡	Wt.‡
A	Hyd. peroxide hydrazine ..	0·22	1,000	0·20	550	0·16	3,450	0·12	1,470	—	—	—	—
B	Oxygen hydrogen..	0·27	170	0·24	100	0·21	590	0·17	325	—	—	—	—
C	„ „ ..	0·29	90	0·27	70	0·23	310	0·18	165	—	—	—	—
D	Fluorine hydrogen	0·24	60	0·22	45	0·19	230	0·15	150	—	—	—	—
E	„ „	0·28	60	0·25	45	0·22	230	0·17	135	0·20	87,000	0·17	9,300
F	„ „	0·29	90	0·27	70	0·23	310	0·18	165	—	—	—	—
G	"X" + "Y" ..	0·23	20	0·22	15	0·17	55	0·14	45	0·16	4,000	0·13	2,000

* These figures assume approx. V2 values for structural efficiency and tank weights, suitably modified for propellant density, and a 20 per cent. improvement in motor specific weight.
† As above, except for a 20 per cent. improvement in structural efficiency and tank weight compared with V2.
‡ As above, except for a 40 per cent. improvement in structural efficiency and tank weight compared with V2.

vary inversely as the propellant density, which is probably true for the second two missions where low accelerations are in order, but unfair to the low density combinations for the orbital case. For this case, a three-step programme was used as a datum, while for the other two, five steps were assumed in each case. Throughout, it was assumed that at least a 20 per cent. improvement in motor specific weight compared with V2 could be realized, the data assumed for V2 being that given in Ref. 13. For the orbital case, calculations were carried out (*a*) assuming no improvement in structural efficiency or specific tank weight and (*b*) assuming that both of these might be improved by 20 per cent. For the other two cases, because of the lower accelerations and consequently stresses involved, it was assumed that this 20 per cent. improvement could be achieved anyway. Initial weights are therefore given on this assumption, and have also been calculated assuming that, by the use of stronger materials and advanced design, as much as 40 per cent. improvement compared with V2 could ultimately be achieved. For each case the calculated optimum values of rocket expansion ratio were assumed, that for the Mars journey being taken to be the same as that for the Moon trip. For this project only the best hydrogen/fluorine and the hypothetical "X" + "Y" reactions were considered worthy of study.

The results of these calculations can be summarized in the following manner, which, in addition, serves as a fairly considered statement of the limited potential of the chemical rocket, based on our present knowledge but assuming reasonable improvements in structural design and motor development.

For the establishment of an orbital rocket, the hydrazine/hydrogen peroxide motor or one of about the same performance, would provide a marginal answer provided great ingenuity was exercised to reduce structure weights to a minimum. Hydrogen and oxygen burnt to give a moderate temperature would make the task comparatively easy, while the use of hydrogen and fluorine would make it easier still. There seems little to be gained by burning these stoichiometrically and the lower temperatures given by a hydrogen-rich

reaction can be used with advantage. If such synthetic propellants as "*X*" and "*Y*" existed, a reduction in take-off weight of nearly 80 per cent. compared with that using hydrogen and oxygen would result.

For the lunar journey, the use of such propellants as hydrazine and hydrogen peroxide is clearly out of the question, while the use of hydrogen and oxygen, even at its optimum exhaust velocity, is marginal. Hydrogen and fluorine result in a more practical solution, and again there seems little advantage in burning these propellants stoichiometrically.

For the Mars project, using an orbital base as a starting point, or for a lunar journey, taking off from the Earth's surface, there are no known propellant combinations which give sufficiently high exhaust velocities. Even using fluorine and hydrogen to give its optimum performance and the best possible estimate of structural efficiency, a quite impracticable mass ratio value is obtained. Even the assumption of a hypothetical synthetic propellant combination fails to result in practical solutions for these particular journeys.

General Conclusions

It would appear from this superficial study that to achieve the best results with chemical rockets, appreciably higher expansion ratios than have been used up to now, will be needed. Associated with these it will probably be necessary to employ higher chamber pressures, in order to keep both the weight and the size of the propulsive units to acceptable values.

The engineering problems resulting from the elevated combustion temperatures of high energy propellants cannot be over-emphasized and it must be remembered that these difficulties will be accentuated by the employment of the higher expansion ratios and pressures which would appear to be necessary. It is quite possible that both of these, for rather different reasons, will be limited by considerations of cooling the combustion chamber.

It is reasonable to expect that the progressive development of existing techniques will permit the use of temperatures perhaps 1,000° C. higher than those already in use. But to exploit to the full the most effective propellants we know of, much more radical development of new methods and the use of new materials will be necessary. The more marginal the use of the chemical rocket becomes, the greater will be the need to reduce all losses to a minimum and to strive for the utmost efficiency in all aspects of design.

Even if we can overcome all the difficulties incurred by their use, the chemical combinations existing in nature which can profitably be used in the rocket motor are very limited. The tasks which can be accomplished with them are correspondingly limited. Of the propellants which have already been used, or could be without a great deal of technical difficulty, only the very best would be powerful enough for a manned venture beyond the atmosphere, and then only for the establishment of an earth satellite vehicle. The most energetic of the known propellants would make a return lunar journey possible if an orbital base were used as a starting point for this. Unless sweeping improvements in structural design were possible, to an extent which does not seem probable, this particular journey would be impracticable using contemporary propellants. There appear to be no propellants at all which would enable a

return journey to be made to the Moon directly from the Earth's surface, or to Mars from an orbital base or to the more remote planets, unless extravagant mass ratios were considered acceptable. It is doubtful whether the advent of synthetic propellants which could be used without great difficulty, would extend the field of usefulness of the chemical rocket appreciably. On the other hand, if they could be made, they might make comparatively easy and practical those tasks which with known chemical propellants are only marginal.

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FABRICATION OF THE ORBITAL VEHICLE

By K. W. Gatland, A. M. Kunesch and A. E. Dixon

(*A paper read to the British Interplanetary Society in London on March* 7, 1953.)

SUMMARY

A space-vehicle using liquid chemical propellant is considered as a means of obtaining a manned reconnaissance of the Moon. The paper discusses methods of assembling this vehicle in a close-satellite orbit from prefabricated parts carried out to the base orbit by 500-tonne freighter rockets. A light jig is recommended to facilitate assembly operations in space. The design suggests an alternative to the method of fuelling by pipe-line, propellant being transferred directly from the supply rockets to the space-vehicle in capsule-like tanks. Consideration is given to possible launching sites for the supply rockets.

Introduction

The history of astronautics is marked by four major turning points. The first was the work of such pioneers as Oberth and Esnault-Pelterie which laid the theoretical basis for space-flight in the 1920's. Secondly was the development of the German A4 rocket, the culmination of original research having its beginning in the early days of the Verein für Raumschiffahrt ev. (VfR) with the Mirak and Repulsor bi-propellant rockets. Thirdly was the recognition of orbital technique as a means of overcoming, to some degree, the limitations of "chemical" propellants by providing a resting and refuelling stage part way up the gravitational slope, and finally, the conception of "ion-drive" which,

in making use of micro-*g* accelerations applied over long periods (and in conjunction with orbital technique) appears to offer a major solution to the problem of interplanetary propulsion.

The present paper, dealing with the basic requirements of an interplanetary vehicle, fabricated in a circumterrestrial orbit, can be regarded as a continuation of the survey we began with our paper "Initial Objectives in Astronautics" prepared in 1950.[1] In that paper, we sought to illustrate the limitations of interplanetary vehicles operating directly from the Earth's surface. It was found that a circumnavigation of the Moon could not be fulfilled even assuming the use of a thermodynamic nuclear reactor with an exhaust velocity of 10 km./sec. Our second paper, or rather three papers (appearing as a symposium under the title "Orbital Rockets" in 1951)[2] dealt with the overall problems of orbital technique and set forth the basic requirements of the various vehicles that we suggested would be involved for the various complementary missions:—

(1) Earth-to-orbit (bringing propellant, materials, supplies and personnel to the terminal, or base orbit, perhaps 800 km. from the Earth's surface);
(2) the orbit-to-orbit vehicle (the true "spaceship," which once established in space would never land on any planet, being left in a circular orbit at both ends of the flight); and finally
(3) the landing craft (a small supplementary vehicle designed for vertical descent with rocket braking, carried to the destination orbit by the parent spaceship).

The course of future development arising from the establishment of small instrumented rockets in circular orbits round the Earth, therefore, appears to be directed towards the orbital space-vehicle as the goal of our present astronautical ambitions.

Considerable publicity has recently been given to certain proposals by Dr. Wernher von Braun, in which he advances the development of a space-programme, largely with military objectives. In the face of these proposals involving Earth-to-orbit 3-step rockets of no less than 6,400 tonnes all-up weight, and inter-orbit vehicles of over 3,600 tonnes, capable of landing 50 men on the Moon, the design studies about to be outlined will inevitably be regarded as "a poor man's von Braun." It should perhaps be explained that our overall conception was arrived at before the appearance of *Collier's* celebrated space-flight issues which originally contained von Braun's proposals, that have since been published in book form by Sidgwick and Jackson.[3] These proposals have received a good deal of criticism, not so much in terms of the actual techniques employed as in the enormous scale of the projects envisaged. As an example, von Braun's plan would expend more than 5,500 tonnes of propellant in each 6,400 freighter rocket which places 33 tonnes of payload in the orbit.

In our reasoning, these proposals are unrealistic as an initial venture and, as we shall see later, expendable freighter rockets with automatic guidance can be designed to bring useful loads to the orbit for an all-up weight of 500 tonnes. These vehicles would be designed for construction on a production basis with

no attempt to recover more than the initial step for re-use. It is further emphasized that there is no need to go to the expense of building a space-station as a pre-requisite of an interplanetary venture, since a high percentage of the structure can be prefabricated on the Earth and "workshop" operations in space virtually eliminated. The personnel engaged on assembling the spaceship from these prefabricated sections would be housed in winged transport rockets, each containing three men, which after a scheduled period of stay in the orbit would descend to the surface and be replaced by further transport craft and fresh crews.

First, then, let us examine more closely the general problems of establishing an orbital space-vehicle.

General Problems

The vehicle we set up will be required to perform an elementary space mission; and as an initial venture we shall specify that it must be capable of making a single journey in a closed elliptical orbit, in which it will circumnavigate the moon, thus permitting close observations of the lunar surface. It will also enable us to carry out proving trials with radio equipment as a necessary preliminary to performing an actual landing.

From this we can determine some initial performance requirements.

On this basis we obtain an impression of the size and weight of the orbital vehicle and hence some idea of the amount of material that must be taken up by freighter rockets into the base orbit.

Economic considerations decree that every attempt be made to utilize material from the freighter rockets in the construction of the orbital vehicle. It will also be necessary to devise a structure for the orbital vehicle that can largely be prefabricated and assembled in the orbit on the principle of "Meccano" construction.

The type of structure envisaged will consist of members, the dimension and weight of which make them easily transportable by the freighter rockets. In addition the individual members must be light and easily handled for assembly purposes. The actual joints between the various members should be as simple as possible and preferably elementary bolt attachments.

As far as possible the main structure should consist of straight members in order to simplify the problem of packing.

Before considering an actual vehicle we shall consider the problem of propellant storage since this will affect the structural design.

Propellant Tanks and Propellant Transfer

Preliminary investigations show that to serve any useful purpose an orbital vehicle must weigh not less than 200 tonnes, although this will depend to some extent on the choice of propellant and power-plant. In any case, it is obvious that the propellant requirements for an orbital space-vehicle cannot be satisfied by refuelling from one tanker rocket.

Briefly, there are three methods of storing propellant in the space-vehicle.

(1) Flexible bags.
(2) A number of small tanks.

(3) A single large tank (which may be subdivided as a protection against total loss of contents in the event of the tank being punctured). Since some form of protection against meteors is desirable, a fourth method might consist of flexible tanks within a light metal container.

If flexible bags are used some preliminary research will be necessary in order to discover a suitable material which will remain flexible whilst in contact with the fuel or oxidant for periods of time which might extend to several months. This would be especially true if the process of squeezing or folding the bags concertina fashion were used to expel the contents.

The second method of propellant storage, namely the use of a number of small tanks, appears to be a fairly logical method. The propellant would be brought up to the space-vehicle in fairly small quantities, either as residual propellant from the main propellant tanks of the tanker rockets or in a special container. Here we have the choice of either pumping the propellant from one vehicle to another or of transferring a completely fuelled tank. In our view, it will be more convenient to transfer the propellant in self-contained tank units.

The shape of the propellant tank not only pre-determines the method of construction but also the way in which the tank is emptied of its contents. Clearly some means of maintaining the propellant in contact with the outlet valve is essential since during periods of free fall, with tanks nearly empty, the residual propellant would break up into globules and difficulty will be experienced in re-starting the motors.

With a cylindrical tank one could use either a flexible bag or a piston which travels towards the valve end under the action of gas pressure.

If one insists on doing without a bag, piston or diaphragm, then the problem of maintaining an adequate fuel supply can be overcome by having in the propellant system a reservoir consisting of a piston-operated tank from which propellant could be drawn during the restarting period. The resultant acceleration would serve to separate the propellants and pressurizing gas in the main tanks and after a suitable period of time the propellant flow from the main supply could be resumed. During the acceleration period the reservoir would be refilled.

Refuelling an orbital vehicle by transfering complete tank units presents mainly problems of handling in the same category as assembly work, and we shall consider this problem later. However, there are several aspects of refuelling by pipe-line which are worthy of consideration.

In order to prevent the propellant from vapourizing during the transfer, the receiving tank must be initially pressurized. As propellant is pumped into the main tank, the gas content will have to be continuously readjusted in order to prevent an excessive pressure rise as the tank is filled. This will be most critical as the tank approaches the full condition.

The initial pressure in the tank must be at least equal to the vapour pressure of the fluid to be pumped in. In the case of liquid oxygen the temperature difference must also be carefully considered.

In the initial stages of fuelling, the heat exchange between the gas and the small quantity of oxidant in the tank may cause the temperature (and hence

the vapour pressure) to rise above the ambient pressure, resulting in considerable losses due to boiling off.

The method of assembly will, of course, depend upon the type and size of structure employed, and the magnitude of the problem can best be assessed by considering an example.

To do this, a particular design of orbital vehicle is considered (Figs. 1 and 5). This was to perform a circumnavigation of the Moon. It was decided to use a multi-tank system. Here the propellant is transported by freight rocket to the orbit in a special container which we shall term a "fuel capsule."

To reduce the amount of pipe work and associated valves, etc., each capsule contains its own gas pressurizing system. The propellant combination chosen was nitric acid/hydrazine. The tanks are cylindrical with hemispherical ends and a piston was adopted for expelling the contents.

Since the time for the mission has been estimated at between one to two weeks which, bearing in mind the time that will elapse between fuelling the capsule and commencement of the mission, makes it obvious that the capsules must be capable of housing their contents for a considerable period. To prevent excessive heating of the tanks, due to solar radiation, a very efficient thermal insulation will be necessary. For this purpose a double skin has been provided and, in addition, the outer shell will serve as an effective "meteor bumper." At the end of the straight portion of the capsule, shallow but substantial rings are provided as hand grips and cable attachment points. The material is welded aluminium. The estimated weight of the capsule is 180 kg. It contains 5 tonnes of propellant, has a diameter of 1·3 metres with a length of 5·2 metres.

The choice of diameter for the capsule is bound up with the design of the freighter rocket. To obtain the desired performance from the freighter rocket it was decided that expendable construction should be employed for the second step thus allowing the orbital step to be enclosed by the annular tanks.

The weight of these tanks is greatly affected by the diameter of the inner wall since this accounts for a substantial proportion of the total weight.

The main problem here is to choose a reasonable length/diameter ratio for the propellant capsule which will also give a reasonable diameter ratio for the surrounding expendable tank.

Preliminary calculations showed that the orbital vehicle would weigh approximately 260 tonnes. It was therefore decided to design for a total of 48 propellant capsules, these being arranged symmetrically round a centre structure and supported by radial members.

There are two tank bays, each consisting of two concentric rings of capsules. Propellant feed pipes lead from these tanks down the support members and inside the centre structure where they join a common manifold leading to the propellant pumps.

The central structure consists of three built-up frames to which the capsule support members are attached. These frames are held in position by a series of longeron members. To ensure that the structure is completely rigid, diagonal members are attached between the longerons.

These longerons and diagonals are triangular shaped members consisting of three equally spaced tubes held in position by diagonal tubular bracing. Both

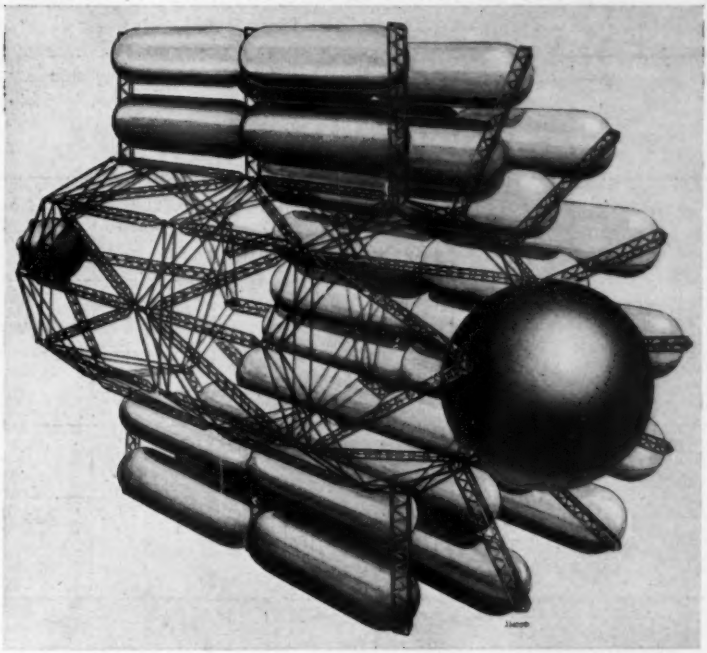

FIG. 1. Inter-orbital space-vehicle for operation between Earth and Moon. Structure will be erected in space from prefabricated girders; propellant capsules brought out in unmanned freighter rockets will be transferred, fully loaded, to the space-vehicle. The spherical cabin would probably be transported in sections; other rocket craft would bring motors and equipment, whilst winged rockets, capable of returning to Earth, bring back the men who perform the assembly operations.

the crew cabin and the rocket motors are mounted on similar triangulated structures.

The structure has been stress analysed and from the results the weights of the various structural elements have been estimated. The weight of a typical longeron member will be approximately 23 kg. and that of a complete frame is estimated at 306 kg.

Method of Assembly

If the problem of assembly rested only on the weight of each member of the structure then we could afford to be light-hearted about this particular scheme. However, due to the large dimensions of the various components and their inherent flexibility, they may well be a little difficult to manoevre into position.

Since we are using only three men the problems of erection make it essential to use an elementary structure which will hold the various members in

position until they can be bolted together. Fig. 2 illustrates how such a jig may be designed. The main frames of the ship are built up using the "net" areas as a jig and then moved into their correct positions and held rigidly until the longerons and diagonals can be attached. For collecting together the

TABLE I

INTER-ORBITAL VEHICLE

(*a*) Technical Data

Dimensions	Length 27·5 metres Diameter 14·6 metres	Approx. duration of supply operation	4 weeks	Weight of permanent structure	5·0
No. of propellant tanks	48	Weight all-up	260·0	Weight of propellant tanks	8·6
Diameter of crew cabin	4·6 metres	Weight propellant (nitric acid/hydrazine)	240·0	Specific impulse	286 secs.
Approx. No. of freighter rockets	53	Weight of crew cabin	4·5	Propellant flow rate kg./sec.	87·5 secs.
No. of manned rockets	3	Payload—3 men, food, water and air* instruments, etc.	1·9	Thrust	25·0

* Allowance of food, water and air for 28 days is at the rate of 6·5 kg./man/day.

(*b*) Operational Sequence

	From Earth orbit	Into lunar orbit	From lunar orbit	Into Earth orbit	Reserve
Velocity required km./second ..	3·06	0·67	0·67	3·06	0·74
Mass ratio ..	2·98	1·27	1·27	2·98	1·3
Initial weight ..	260·0	80·93	62·7	49·3	16·5
Final weight ..	87·1	63·73	49·4	16·5	12·7
Weight of propellant used ..	172·9	17·2	13·3	32·8	3·8
No. of tanks jettisoned after each manoeuvre and weight	34 = 6·1 tonnes	4 = 0·71 tonnes	—	—	—
Firing time ..	33 mins. 3 secs.	3 mins. 20 secs.	2 mins. 35 secs.	6 mins. 22 secs.	44 secs.

Note:—All weights are in tonnes.

component parts as they arrive in the base orbit a light winch would be employed. Naturally, once these materials have been "captured" they will be retained at the point of assembly by anchor lines.

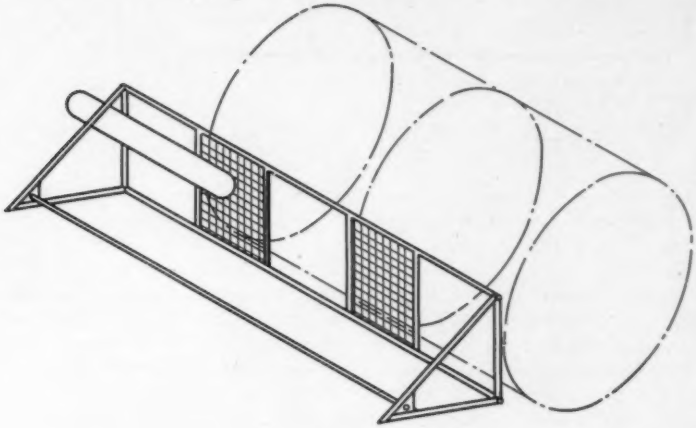

FIG. 2.

Ground to Orbit Vehicles

The freighter rocket shown in Fig. 3 has been designed as a three-step vehicle, to carry a payload of five tonnes (propellant or material) into an 800 km. orbit, for the construction of an inter-orbital spaceship designed to circumnavigate the Moon.

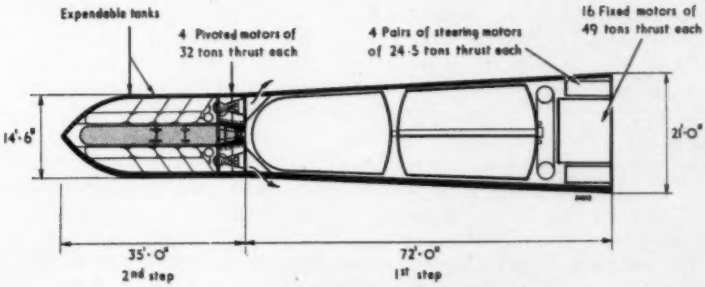

FIG. 3.

The first and second steps use liquid oxygen and hydrazine as propellant whilst nitric acid and hydrazine are employed for the third step. The payload capsule (Fig. 4) the dimensions of which determine the final configuration of this step is, as previously mentioned, 1·3 metres in diameter and 5·2 metres long with hemispherical ends.

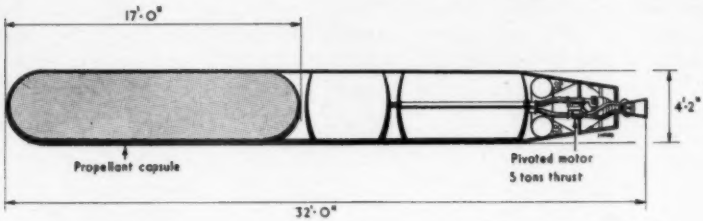

FIG. 4.

During the initial ascent through the atmosphere the third step is contained within the expendable tanks of the second step and is thus protected from any possibility of frictional heating.

The initial weight of the final step is 12·0 tonnes and in view of the fact that the take-off trajectory, over the period of firing subsequent to jettisoning the second step, is inclined at only a few degrees from the horizontal, the thrust has been restricted to 5 tonnes, with a total burning period of nearly six minutes.

The second step, using expendable construction giving 4 propellant tank bays, 3 of which are jettisoned as they are emptied, has a central "sleeve" with a detachable nose, into which is fitted the third step.

The initial weight of the second step is 75 tonnes, and to carry this, the first step, which is of conventional (i.e., non-expendable) design, weighs 435 tonnes; this gives an all-up weight at take-off of 510 tonnes and an effective mass ratio of 24.

A certain number of the freighter rockets will be launched with structural members contained in the payload capsule in place of propellant and difficulties will be experienced in making up an optimum load due to the bulk density ratio of these members. This, however, would make no difference to the economic design since the deficiency would be made up by the fact that propellant would be left over in the rocket's tanks and this would be used as a reserve store in the orbit. An "overload," however, if required, is not quite so conveniently accommodated and modifications to the design would be required to improve performance. To a certain extent, these modifications could best be carried out by the addition of further tank bays to the second step. For example, if the payload is increased to 6 tonnes an additional tank weighing 22 tonnes could be added to the second step. This would increase the effective mass ratio from 3·4 to 4·13 and so counteract the reduced mass ratios of the first and third steps.

For the Earth return vehicle a considerably modified version of the first and second steps would be required, together with a high speed glider of delta planform designed to descend from the orbit, using rocket braking, and a combination of drag drogues or dive brakes to retard its velocity in the upper atmosphere.

A survey of the problems involved in using this descent technique was given in a paper presented by Mr. T. R. Nonweiler at the International Astronautical Congress in 1951.[4]

TABLE II

TECHNICAL DATA

EARTH TO ORBIT FREIGHTER ROCKET

	1st Step	2nd Step 3 expendable tanks	3rd Step
Payload	76·0	12·3	5·22
Structure inc. motors and pumps	74·3	12·2	0·8
H_2O_2 and catalyst ..	21·7	3·1	0·28
Propellant	348·0 Liquid oxygen/ hydrazine	48·4 Liquid oxygen/ hydrazine	6·0 Nitric acid/ hydrazine
All-up weight	520·0	76·0	12·3
Total propellant flow rate in kg./sec.	2,940·0	441·0	17·5
Thrust	900·0	135·0	5·0
Effective specific impulse	306·0 secs.	306·0 secs.	286·0 secs.
Characteristic velocity, km./sec.	3·72	3·69	2·0 (overall 9·41)
Mass ratios	3·46	3·1 actual 3·41 effective	2·04 (overall 24·0)
Firing times	124–126 secs.	117 secs.	358 secs. (overall 600 secs.)
Dimensions in metres ..	32·5 m. long × 6·4 dia.	10·7 long × 4·4 m. dia.	9·75 m. long × 1·28 m. dia.
Max. acceleration ..	6 g.	6·9 g.	0·8 g.
No. of motors	16 fixed 8 pivoted	4 pivoted	1 pivoted

Note:—All weights are in tonnes.

A return glider with a landing weight of approx. 5 to 6 tons carrying a half ton of payload, would probably require a second step of conventional rather than expendable design since the latter form of construction requires the third step to be contained within the hull of the second step, as in the freighter rocket (Fig. 3). Securing the expendable tank bays with the wings of the glider separating the half tanks, would present a considerable problem. In addition the large wing area located so far forward of the centre of gravity might create difficulties at take-off and during the initial ascent due to aerodynamic instability and to counteract this fins may have to be added to the first step.

The increased structural weight of the first and third step coupled with the reduced performance of the second (conventional) step would mean an increase in the take-off mass of approximately 200 tonnes, which gives a launching weight of 740 tonnes.

In selecting a suitable launching site, a number of factors have to be borne in mind:—

1. Range and points of impact of jettisoned tanks and steps.

2. If, as is probable, the launching site is so located that boosters, etc., will fall back into the sea, the track of the main shipping routes should not lie anywhere near the dropping areas.

3. Access to site, whether by sea or by rail, should be easily possible in view of the large quantities of materials required.

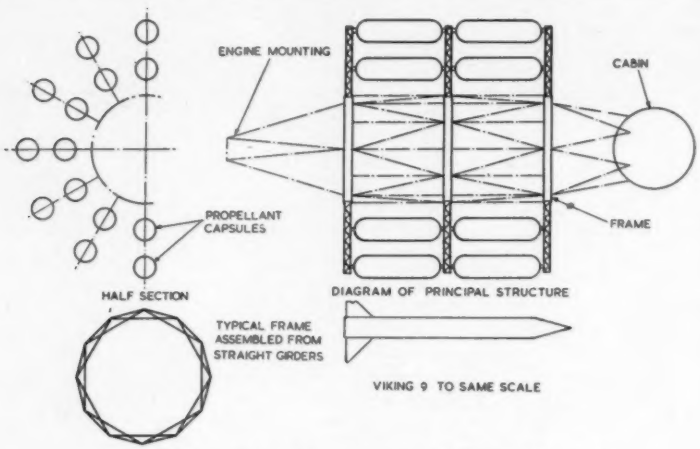

FIG. 5. Diagram of general arrangement of Inter-Orbital vehicle.

4. Minimum inclination of orbit relative to the plane of the Equator, to obtain maximum benefit from the Earth's rotation.

5. Weather conditions, which may rule out a site that otherwise would be quite suitable.

When it is borne in mind that the 12½ tonne V2 rocket took several hours to prepare for launching, the problems of fuelling and servicing a 500 tonne, three-step, freighter rocket can be appreciated. The operation would probably take at least 24 hours to complete and postponement due to the sudden emergency of bad weather would be costly and possibly dangerous. In the case of the operational use of V2, the rocket was fuelled at the launching site and fired the same day. If, for some reason, it could not be fired within two to three days, the likelihood of a failure occurring was greatly increased, as the Germans found to their cost.

6. Owing to the probability, when launching large numbers of these satellite rockets, that one or more will go seriously off-course due to some defect, automatic cut-off will have to be employed; this will prevent any possibility of the complete rocket (still weighing several hundred tonnes), or of the jettisoned steps, falling on to a populated area. For this reason, it would be advisable to clear an area of 80 km. radius from the site, on to which any rocket malfunctioning at an early stage could be ditched. In the case of a failure in the third step, however, little could be done unless it occurred very soon after separation from the second step, as the range of the step could vary from approximately 2,700 km. (in the case of failure occurring shortly after leaving the second step) to the extreme case in which the motor fails at the point of reaching the transfer orbit, when (due to the fact that the velocity increment could not be added for transferring to the final 800 km. orbit) the step would continue to circumnavigate the Earth in its elliptical transfer orbit

undergoing a fractional loss of height each time it "grazed" through the upper fringes of the atmosphere at its nearest approach. This would continue until eventually the rocket encountered denser air, when due to its comparatively light construction and lack of streamlined form, it would undergo violent deceleration and heating effects which would destroy it. The debris would come down as a shower of pieces spread over a wide area, the motor, pump and miscellaneous air bottles being the most dangerous items.

Island launching sites in the Pacific, particularly on the Christmas Island group, would be high on the list of suitability with some 7,200 km. of ocean extending to the Galapagos Islands. These islands are located near the Equator, approximately 2° North latitude, and the firing line would cross no major shipping routes. However, due to the vast distances in the Pacific, the material supply lines would be rather extended.

A position on the East African coast in Kenya would give an adequate firing lane extending across the Indian Ocean to Indonesia. The supply position would be better than for the previous example but the dropping area would straddle a number of shipping routes.

Brazil would present some possible sites on or near the Equator with a clear stretch of the South Atlantic, extending into the Gulf of Guinea, in which to drop tanks and steps.

A large and highly trained organization will be required, with extensive radar tracking coverage, and capable of operating with a high degree of accuracy. The launching techniques will follow closely those already used for existing high-altitude rockets.

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1953 CHRISTMAS CARDS

Orders may now be placed for the Society's Christmas Card for 1953 which has been painted by Mr. J. W. Wood from a design by Messrs. K. W. Gatland and A. M. Kunesch.

The illustration, reproduced from the author's book, *Space Travel*, shows a spaceship with a thermodynamic nuclear reactor taking aboard propellant in a satellite orbit, preparatory to its departure for the Moon.

We hope that all members will make use of this most attractive card, which not only helps to solve a universal Christmas problem, but also serves to spread the idea of astronautics.

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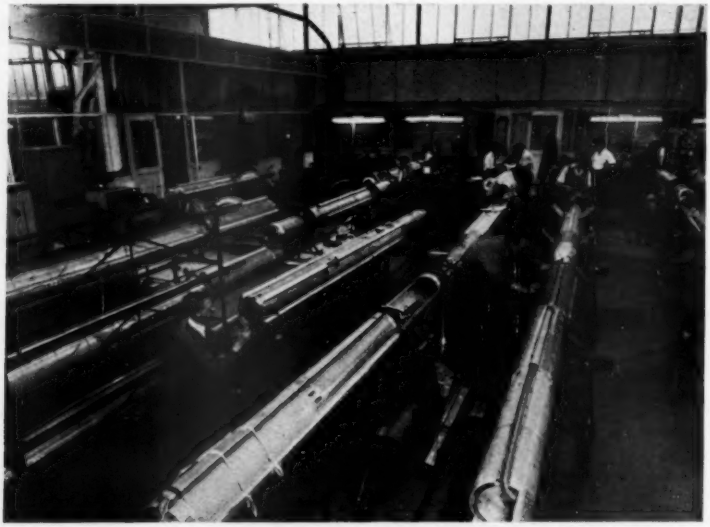
Guilded Missile with eight Auxiliary Boost Motors
Guided Rocket accelerating after launching assisted by auxiliary Boost Motors



Corner of a rocket assembly shop at British Ministry of Supply Establishment

PROGRESS OF BRITISH GUIDED ROCKETS

The Minister of Supply, Mr. Duncan Sandys, issued the following statement on leaving for Australia on August 24, 1953.

Australian Co-operation

One of the matters which I shall be discussing with the Australian Government is the progress on our joint efforts on the development of guided rockets. This is in every sense a combined operation, and is an outstanding example of effective co-operation between fellow members of the British Commonwealth.

The development of these rocket weapons is carried out initially in Britain, by the experimental establishments of the Ministry of Supply, or by outside firms in industry. After a weapon has undergone preliminary firing tests here, it is sent to Australia, where full-scale trials with explosive warheads can be carried out over land on the giant rocket range at Woomera. If in the light of these trials modifications are needed, they can for the most part be carried out in the extensive engineering and electronic workshops, which have been set up in the vicinity of the range by the Australian Government and by the Australian branches of the principal guided rocket firms in Britain.

During the past year our scientists and technicians have made further remarkable advances in the development of guided rockets to meet the various requirements of the three Fighting Services. Their work has centred upon two main problems—speed and guidance.

Supersonic Speeds

Speed is of particularly vital importance. In order to tackle bombers flying as fast as or even faster than sound, and at heights above 50,000 ft., our rockets have to travel at more than 2,000 miles an hour. In achieving this prodigious performance, many technical problems of great complexity have had to be solved.

If it is to be able to intercept and destroy an approaching raider before it can release its bombs, the guided rocket must lose no time in building up its velocity. This has made it necessary to fit the rocket with auxiliary boost motors, which rapidly accelerate it up to its full supersonic cruising speed. Having served their purpose, the auxiliary motors are automatically discarded. To make these boosts drop off simultaneously and smoothly, without interfering with the course of the missile, has presented many engineering and aerodynamic difficulties which have, however, been successfully overcome.

Manoeuvrability

Anti-aircraft rockets must not only travel at high speed, but must also be able to change course rapidly if the enemy bomber takes evading action. I can say with confidence that no piloted aeroplane could hope to out-manoeuvre guided rockets of the types we are now developing. They are capable of high-speed twists and turns, which create such intense strains and pressures as neither the human body nor the wings of any aircraft could withstand.

Guided Missile with eight Auxiliary Boost Motors

Four Twin Boost Motors fall away, while the rocket carries on under its own power at a speed of about 2,000 miles per hour

(*Other photographs of British Guided Missiles appeared on page* 274 *of the November,* 1952 *"Journal" in connection with an article, "Progress in Guided Rockets"*)

Since these stresses increase with every extra pound carried, continuous efforts are being made to reduce weight through the use of lighter materials. New methods have been evolved for moulding large plastic structures, and before very long it may be possible to make the whole of the rocket's casing out of these light materials instead of metal. This will not only reduce weight without loss of strength, but will also reduce cost—a matter of great importance.

Guidance Mechanism

The most complex component of a guided rocket is its steering mechanism—a kind of electronic brain. In some cases this may contain 10 times as many valves as a large television set, all of which have to be fitted into a missile a few inches in diameter, and have to be made robust enough to withstand the shock of firing.

Some types of rockets are known as "beam-riders." They fly up a radar beam which follows the target automatically and is operated from the ground. Even at long ranges these rockets can keep themselves within a few feet of the centre of the beam.

Then there are so-called "homing-rockets." As soon as they have been launched, these missiles take over control completely. They lock their guidance mechanism on to the enemy plane and, without any further assistance from the ground, they steer towards it, changing course as necessary. Unlike ground-controlled "beam-riders," their capacity is unaffected by distance. In fact, the nearer they approach the target the stronger its "scent" becomes, and the easier it is to "home" on to it.

Rockets for other Roles

In addition to missiles fired from the ground or from ships, we have reached an advanced stage in the development of guided rockets to be launched from fighter aircraft. These will increase the killing power of our fighters many times over, and will enable them to engage an enemy bomber from a distance beyond the range at which it can defend itself with any conventional aircraft gun.

Initially our efforts were concentrated on the problem of defence against enemy air attack. For this purpose a series of missiles has been evolved—some to be launched from the ground, some from ships and some from fighter planes. Whilst these anti-aircraft weapons will be the first guided rockets to be brought into Service, they will be followed by other types for use in various artillery and bombardment roles.

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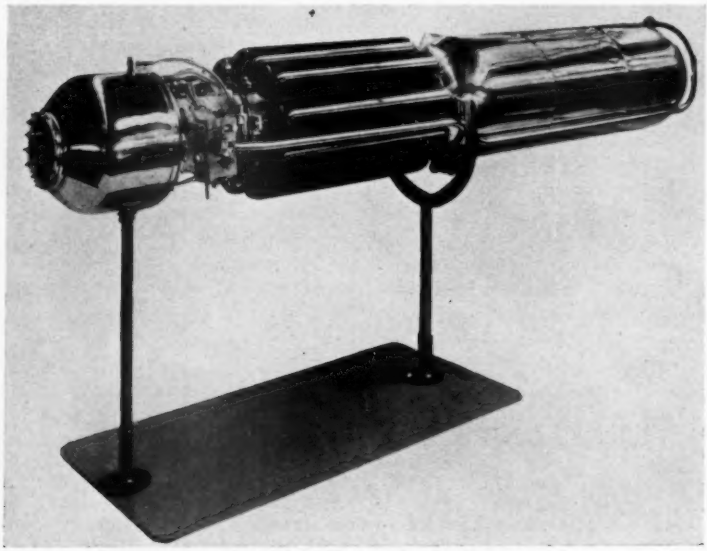
THE DE HAVILLAND SUPER SPRITE

A new assisted-take-off rocket motor

Following quickly on the announcement of the de Havilland Gyron engine, which, as revealed on July 30, 1953, is now giving on test a substantially greater thrust than announced for any other jet engine, comes the news of the de Havilland Super Sprite, a new assisted-take-off rocket motor designed in the first place for use with military aircraft. Details just released reveal that the Super Sprite, in its present form, gives a total impulse of 120,000 lb.-seconds. It weights 600 lb. and has a duration of about 40 seconds with a maximum thrust of 4,000 lb.

The modern high-performance aircraft, whether jet or propeller driven, is capable of carrying in the air a far greater load than it is able to lift off the ground even from the longest runways. The availability of a rocket motor, with a power-weight ratio greatly in excess of that of conventional power plants, providing additional power during the critical period, largely overcomes this handicap and enables the aircraft to take off safely with greatly increased loads.



The De Havilland Super Sprite

The Super Sprite employs hydrogen peroxide with kerosine or petrol injection and an important new feature is the solid catalyst which replaces the liquid calcium permanganate used in the original Sprite rocket motor. The

purpose of the catalyst is to break down the hydrogen peroxide into steam and free oxygen. It is this conversion of liquid peroxide into gaseous steam and oxygen that provides the bulk of the thrust of the motor. The injection of kerosine or petrol fuel and its combustion, which is made possible by the presence of the free oxygen in the exhaust stream, increases the energy exerted by the rocket, and this additional energy can be used either to increase the thrust or to extend the duration of the run. The kerosine or petrol used in the Super Sprite is similar to that used in the main engines of the aircraft, and it is this factor which dictates the choice between the alternative fuels, both of which are equally suitable for injection in the rocket. Hydrogen peroxide has a marked advantage over other possible fluids which may be used in rocket propulsion, such as nitric acid or liquid oxygen, in that it is safe and easy to handle and has good long-storage characteristics.

The introduction of the solid catalyst eliminates the need for a separate permanganate tank and leads to a simplification in the design and greater ease of handling. It also has the advantage of eliminating the "brown steam" exhaust, a characteristic of rockets using permanganate. The exhaust jet of the new unit is completely smokeless.

Structural details of the Super Sprite or of the types of aircraft for which it has been designed are not revealed, nor is it permissible to discuss the future development of this and other rocket-motor projects now being undertaken by the de Havilland Engine Company.

NOTES AND NEWS

Dr. Günter Loeser—Obituary

It was with the utmost sorrow that the delegates to the Fourth I.A.F. Congress learned of the death, only a few days before the opening of the Congress, of Dr. Günter Loeser, Vice-President of the Federation. Dr. Loeser had been indefatigable in working for co-operation between national societies, and it had been a source of great regret to him that an Air Force meteorological research project of which he was director prevented him from being present at Zurich.

It was on this work that he was killed, at O'Neill, Nebraska, in a helicopter crash.

Astronautics has suffered a severe loss through Dr. Loeser's death, and as a memorial arrangements are now being made for the inauguration of a "Günter Loeser Medal," details of which will be announced by the I.A.F. later this year.

Change in Midlands Branch Lecture Programme

The speaker at the Midlands Branch Meeting on November 7, 1953, will be Mr. A. V. Cleaver, F.R.Ae.S., who will speak on "A Programme for Achieving Interplanetary Flight," in place of Mr. Coombes, who was advertised to read his paper on "Considerations in the Design of Rocket Structures."

Mr. Cleaver's paper will discuss the various phases of research and development which need to be covered before interplanetary flight can be achieved.

Dr. L. R. Shepherd (right) presents the Hermann Oberth Medal to Mr. A. V. Cleaver at the Society's Meeting on September 12, 1953

Presentation to A. V. Cleaver

A very pleasing ceremony took place at the London meeting on August 29, 1953, when Dr. L. R. Shepherd presented Mr. A. V. Cleaver with the Hermann Oberth Medal, awarded by the *Gesellschaft fur Weltraumforschung* in recognition of his outstanding contributions to astronautics and rocketry.

Marriage of Mr. A. C. Clarke

We know that all members of the Society will join with us in extending our very good wishes to our Chairman, Mr. A. C. Clarke, on the occasion of his marriage to Miss Marilyn Mayfield of Jacksonville, Florida, U.S.A. The wedding took place in New York on June 15, 1953.

School Astronautical Society

Representative of a number of similar groups is the Manchester Senior Boys School Astronautical Society, formed by Mr. J. H. Holt, a Fellow of the B.I.S. This Society recently held an exhibition of photographs, drawings and models, many of which were lent or given by B.I.S., B.A.A., the Palomar, Wilson, Yerkes, Lick and Belgrade Observatories, also the White Sands Proving Ground. About 300 people visited the exhibition.

The Society was formed at the request of the boys after visits to the Godlee Observatory. Its activities include further visits to other scientific establishments in and around Manchester, observations of planetary bodies, lectures and discussions, and the collection of a library.

Book List No. 11

Copies of the latest Book List are now available to all members of the Society on application to the Secretary.

Lending Library

The Lending Library is open only to *Fellows of the Society resident in Great Britain* and is operated from 12, Bessborough Gardens, London, S.W.1.

A list of books available can be obtained on demand.

Erratum

It is regretted that the introductory paragraph to the article by Mr. Ross, "*Is Astronautics Escapism?*" (*J.B.I.S.*, **12** (4), 175, July, 1953), was inadvertently omitted.

This paragraph reads as follows:—

"Why bother with other worlds while so much remains to be put right *here* . . . that is probably what people are thinking when they label astronautics *Escapism*."

"That's handy, Bert!"

Reproduced by courtesy of the Editor of the "Daily Sketch."

ABSTRACTS

Edited by J. HUMPHRIES

Abbreviations of titles of journals were given in the May, 1950, issue of the *Journal*, and addenda have appeared in subsequent issues. The following is a further addendum to the list:—

Cienc. y Tecn.	Ciencia y Tecnica.

Many of the articles noted are available on loan to members resident in the British Isles. Requests should be addressed to J. Humphries, 97, Churchill Avenue, Southcourt, Aylesbury, Bucks.

AERODYNAMICS

(257) **Nozzle characteristics in high-vacuum flows. Rarified gas dynamics.** R. G. FOLSOM. *Univ. of Calif., Berkeley, Rept. HE*-150-102, 5 pp. (*Sept.* 10, 1952).

AIRCRAFT

(258) **The dawn of aviation's new era.** D. G. SAMARAS. *Soc. Automot. Engrs. Prepr. No.* 26, 18 pp. (*Jan.*, 1953).

Review of possible future methods of propulsion and problems associated with extremely high-speed flight.

(259) **Astronautical contributions.** *Alata*, **9** (5), 33-34 (*May*, 1953). (*In Italian.*)

Abstracts of some of the papers delivered at the 3rd International Congress on Astronautics.

(260) **Rockets breathe new life into autogyro.** D. A. ANDERTON. *Aviation Wk.*, **58** (25), 28-30 (*June* 22, 1953).

Use of rocket units to rotate blades for take-off and ordinary propeller for horizontal flight. Also one man autogyro with rockets only of total weight 65 lb.

ASTRONAUTICS

(261) **Problems of space travel.** *Science News Letter*, **62**, 180 (*Sept.* 20, 1952).

(262) **Vehicle for space.** J. G. VAETH. *Flying*, **51**, 26 (*Oct.*, 1952).

(263) **Space superiority.** W. VON BRAUN. *Ordnance*, **37**, (197), 770-775 (*March-April*, 1953).

Review of the military use of space stations—as observation posts and as missile bases. Also discusses vulnerability and concludes that the chance of destruction is low.

(264) **Navigational calculations in space flight. (Investigation of the effect of the precision of astronomical data.) Pt. I. Planetary orbital distances.** H. B. KETCHUM. *J. Space Flight*, **5**, (4), 1-8 (*April*, 1953).

ASTRONOMY

(265) **The temperature of Jupiter.** D. ALTER. *Griffith Observer*, **17**, (2), 19-22 (*Feb.*, 1953).

(266) **The canals of Mars.** T. R. CAVE. *Griffith Observer*, **17**, (2), 14-18 (*Feb.*, 1953).

(267) **The Moon.** **1**, (4) (*May*, 1953).

Contains various lunar observations.

BIOLOGY AND MEDICINE

(268) **Animal studies of the subgravity state during rocket flight.** J. P. HENREY, E. R. BALLINGER, P. J. MAHER and D. G. SIMONS. *J. Aviation Medicine*, **23**, 421-432 (*Oct.*, 1952).

(269) **The biological effect of cosmic rays.** J. EUGSTER. *Natur u. Tech.*, **7**, (5), 135-137 (*May*, 1953). (*In German.*)

Shortened version of paper delivered at 3rd International Congress on Astronautics.

CHEMISTRY

(270) **Apparatus for metering and distributing anhydrous ammonia in a liquid state.** T. L. BAGGETTE and A. R. COPITHORNE. *U.S. Pat. No.* 2,612,760 (*Oct.* 7, 1952).

PROJECTILES

(271) **More power for guided missiles.** *Business Wk.*, 120 (*Sept.* 13, 1952).

(272) **The concept of missile weight.** H. HABER. *Aero Dig.*, **66**, (4), 86-88 (*April*, 1953).

Suggestion to use summation of gravitational force and inertial force on a body as its "true" weight.

(273) **Ing. Aurelio Robotti tests a new type of liquid propellant rocket.** *Alata*, **9**, (5), 26-27 (*May*, 1953). (*In Italian.*)

(274) **Recent developments in rockets and guided missiles in the United States.** N. J. BOWMAN. *J. Space Flight*, **5**, (5), 1-9 (*May*, 1953).

Discusses and tabulates post-war U.S. rockets.

ROCKET MOTORS

(275) **Practical development of rocket motors.** R. DYRGALLA. *Cienc. y Tecn.*, **119** (602), 39-58 (*Aug.*, 1952). (*In Spanish.*)

Review of theoretical and empirical relationships involved in the design of rocket engines.

(276) **Rocket motor bodies cold-formed on a single press.** R. E. HANSEN. *Machinery*, **59**, 200-203 (*Oct.*, 1952).

Production of 3.5 in. bazooka body and venturi.

(277) **Nozzle structure for effecting striated flow of combustion gases.** R. H. GODDARD. *U.S. Pat. No.* 2,612,746 (*Oct.* 7, 1952).

(278) **Rotatable combustion chamber.** R. H. GODDARD. *U.S. Pat. No.* 2,612,750 (*Oct.* 7, 1952).

(279) **Feeding apparatus, including injectors adapted to supply combustion liquids under pressure to a combustion chamber.** R. H. GODDARD. *U.S. Pat. No.* 2,612,752 (*Oct.* 7, 1952).

(280) **Rocket having adjustable discharge passage.** L. A. SKINNER. *U.S. Pat. No.* 2,612,747 (*Oct.* 7, 1952).

(281) **Rocket motor cooling system.** R. C. TRUAX. *U.S. Pat. No.* 2,625,007 (*Jan.* 13, 1953).

Annual Report
of the
British Interplanetary Society
(Limited by Guarantee)

EIGHTH ANNUAL GENERAL MEETING

NOTICE IS HEREBY GIVEN that the EIGHTH ANNUAL GENERAL MEETING of the BRITISH INTERPLANETARY SOCIETY will be held in the Kent Room, Caxton Hall, Caxton Street, London, S.W.1, on **Saturday, the 5th day of December, 1953,** at 6 o'clock in the evening precisely, for the transaction of the business specified in the Agenda below.

AGENDA

1. To hear the Chairman's speech.
2. To receive the annual statement of accounts and balance sheet and the auditor's report thereon.
3. To consider and, if thought fit, to pass as a SPECIAL RESOLUTION, "That Article 30 be amended to read as follows:—

   'Any group of members in one area may, subject to the sanction of the Council, form a Branch and elect a Branch Committee to act within the limits imposed by the Council.' "
4. To elect four Members of the Council of the Society. In accordance with the provisions of Article 15, the following four Members of the Council will retire at that meeting and, with the exception of Mr. A. C. Clarke, offer themselves for re-election:

   E. Burgess. A. E. Slater. R. A. Smith.

   The following additional names have also been nominated:—

   D. J. Cashmore *Fellow.* D. Hurden *Fellow.*
   C. A. Cross *Fellow.* D. F. Lawden *Fellow.*

   Since the number of nominations exceeds the number of vacancies, election will be by postal ballot. The necessary ballot papers are enclosed with this agenda.
5. General discussion of the affairs of the Society during the past year.
6. Any other business.

By order of the Council,

L. J. CARTER,
Secretary.

A member who cannot be personally present at the meeting may appoint by proxy some other person (who must be a member of the Society) to attend and vote on his behalf, subject, however, to the limitation that a proxy cannot vote except on a poll.

The British Interplanetary Society

Limited (by Guarantee)

INCOME AND EXPENDITURE ACCOUNT for the year ended 30th September, 1953.

1952	*Expenditure.*	£ s. d.	£ s. d.
£			
	To Office and Management Expenses—		
105	Rent, Light, Heat, Cleaning and Insurance	219 3 11	
197	Salaries and National Insurance	348 9 6	
335	Printing and Stationery ..	275 3 6	
279	Postage, Telegrams and Telephone	354 1 1	
93	Publicity and Advertising ..	122 0 0	
70	Travelling and Entertaining Expenses	109 5 3	
48	Library Expenses	65 12 0	
10	Accountancy and Audit ..	12 12 0	
12	Bank Charges and Differences on Exchange	13 18 7	
134	Repairs to Premises	— — —	
60	Sundry Expenses	25 14 9	
			1,546 0 7
1,723	" **Publications**		2,362 19 10
170	" **Expenses of Meetings** ..		216 16 4
	" **International Astronautical Federation—**		163 9 7
167	Contribution and Expenses ..		
	" **Depreciation and Amounts written off—**		
85	Furniture and Equipment ..	282 1 7	
113	Library	112 5 4	
			394 6 11
300	" **Transfer to Subscription Reserve**		200 0 0
619	" **Excess of Income over Expenditure** for year carried to Accumulated Fund		457 16 11
4,520			£5,341 10 2

1952	*Income.*	£ s. d.
£		
3,424	**By Membership Subscriptions**	4,127 18 2
415	" **Entrance Fees**	403 4 0
404	" **Donations**	296 5 9
218	" **Library Subscriptions and Sale of Journals**	398 10 9
59	" **Sundry Receipts**	115 11 6
4,520		£5,341 10 2

BALANCE SHEET as at 30th September, 1953.

1952 £	*Capital and Reserves.*	£ s. d.	£ s. d.
	Accumulated Fund—		
382	Balance at 30th September, 1952	1001 13 1	
	Add Excess of Income over Expenditure for year to 30th September, 1953	457 16 11	
619			1,459 10 0
	Compounded Subscriptions		
755	**Reserve Account**		1,035 0 0
500	**Subscriptions Reserve Account**		700 0 0
2,256	Total of Capital and Reserves		3,194 10 0
	Current Liabilities.		
170	**Subscriptions paid in advance**	167 1 7	
22	**Sundry Creditors**	207 1 10	
			374 3 5
2,448			£3,568 13 5

Certified to be a true copy—

A. C. CLARKE, G. V. E. THOMPSON } *Members of the Council.*

L. J. CARTER, *Secretary.*

1952 £	*Fixed Assets.*	£ s. d.	£ s. d.
	Furniture and Equipment—		
821	At Cost	1033 6 6	
—	*Less* Items written off	109 8 3	
821		923 18 3	
151	*Less* Depreciation	323 18 3	
670			600 0 0
	Library—		
390	At Cost	502 11 11	
50	*Less* Items written off	52 11 11	
340		450 0 0	
190	*Less* Depreciation	250 0 0	
150			200 0 0
820	Total of fixed assets		800 0 0
	Current Assets.		
46	**Sundry Debtors**	51 7 6	
33	**Stock of Postcards, etc.**	— — —	
78	**Payments made in Advance**	92 7 9	
809	**Establishment Fund Deposit Account**	1100 0 0	
645	**Cash at Bank Current Account**	1490 12 7	
17	**Cash in Hand**	34 5 7	
			2,768 13 5
2,448			£3,568 13 5

REPORT OF THE AUDITOR TO THE MEMBERS OF THE BRITISH INTERPLANETARY SOCIETY

I have obtained all the information and explanations which to the best of my knowledge and belief were necessary for the purposes of my audit. In my opinion proper books of account have been kept by the Society so far as appears from my examination of those books. I have examined the above Balance Sheet and annexed Income and Expenditure Account which are in agreement with the books of account. In my opinion and to the best of my information and according to the explanations given to me, the said accounts give the information required by the Companies Act, 1948, in the manner so required, and the Balance Sheet gives a true and fair view of the state of the Society's affairs as at 30th September, 1953, and the Income and Expenditure Account gives a true and fair view of the surplus of income over expenditure for the year ended on that date.

35, Brian Avenue,
Sandershead, Surrey.
1st November, 1953.

T. W. WOODS,
Incorporated Accountant.

EIGHTH ANNUAL GENERAL MEETING

(For the convenience of members unable to be present, the Chairman's speech is reproduced below.)

CHAIRMAN'S ADDRESS

GENTLEMEN,

Before presenting my report, I would like to remind you that this year marks the twentieth anniversary of the founding of the Society. Next year we will come of age, and I cannot help thinking that our founders must be surprised and pleased to see the growth in prestige and numbers that has taken place since we began our existence in Liverpool back in 1933.

The past year has again been a period of steady expansion, and we are settling down in our new office. Having a permanent address and telephone number has been an enormous advantage, and we now find it hard to imagine how we managed to run the Society in the old days!

Membership

Our membership at September 30, 1953, was 2,419, comprising 584 Fellows and 1,835 Members, an increase of 409 during the year. The figures for the last five years are now:—

Total	Membership	at	September	30,	1949	=	577		
"	"	"	"	"	1950	=	925	(Annual increase,	60%)
"	"	"	"	"	1951	=	1,491	(" "	61%)
"	"	"	"	"	1952	=	2,010	(" "	35%)
"	"	"	"	"	1953	=	2,419	(" "	20%)

These figures still give cause for satisfaction, but we are rather disturbed by a trend which they do not indicate. This is the large number of members who do not renew their subscriptions after a relatively short period of membership. Since the beginning of 1946, in fact, more than a thousand members have joined us and then left again. We do not know what can be done about this: perhaps the enthusiasm of these members waned when they discovered space-flight was not likely to be achieved in the immediate future, and that the B.I.S. was primarily a technical organization.

Lectures and Meetings

The Society held 7 general meetings in London (including the Annual General Meeting on December 5, 1952), as well as 8 meetings in Manchester and 7 meetings in Birmingham. A number of meetings were also held by the Provisional Yorkshire and Western Branches in Leeds, Sheffield and Bristol.

In addition, two informal meetings took place in London so that members could get together and talk with each other and Members of the Council. The first of these was on (appropriately enough!) November 5, 1952, at the *White Horse*, Fetter Lane, E.C.4, while the second took place on August 19, 1953, at the *Mason's Arms*, Maddox Street, W.C.1—a pre-war meeting place of the Society. On the latter occasion, we were very pleased to have with

us Mr. and Mrs. Haley and Mr. and Mrs. Durant of the American Rocket Society.

Eight members of the Society, including Dr. Shepherd, Mr. Carter and myself, were able to attend the Fourth I.A.F. Congress in Zurich, an account of which is given elsewhere in this issue.

Of great significance also was the Royal Society's conference on rocket exploration of the upper atmosphere, held at Oxford during August 24 to 26. This was also well attended by members of the B.I.S.

On March 14, 1953, your Chairman had the pleasure of lecturing to the *Nederlandse Vereniging voor Ruimtevaart* at the Hague, and was shown many of the Dutch places of interest by that Society's President, Professor Kooy.

An interesting and important outside event during the year was a visit of the Society, arranged by the North-West Branch by courtesy of Professor A. C. B. Lovell, to the radio astronomy experimental station at Jodrell Bank.

Once again, members of the Society have given innumerable lectures to outside bodies, but even so it has proved impossible to comply with all the requests received. To assist members willing to undertake such lectures a set of lecture notes was made generally available in September, 1953, and we have now prepared a film strip containing fifty-two illustrations of astronautical interest which will be of great value for lecturers. The strip may be obtained from the Secretary at the modest price of 10s., and even those members who are not interested in lecturing may like to have a copy for their own libraries.

We are now endeavouring to increase our panel of lecturers, and we would particularly like to hear from Fellows resident in the London area who may be willing to speak to Youth Clubs, Rotary Clubs, etc. Being on the panel involves no obligation: arrangements are made directly between the lecturer and the organization concerned, the B.I.S. merely providing the list of possible speakers.

Publications

During the past year we have issued the following publications:—

(*a*) Six *Journals*, viz. Vol. 11, No. 6 (November, 1952) and Vol. 12, Nos. 1 to 5 (January, March, May and July and September, 1953).
(*b*) Tenth and eleventh printing of the Book List, which continually expands, and is always available to members on request.
(*c*) List of Members correct to 30th June, in September, 1953.
(*d*) Annual Report in October, 1952.

In addition, those Fellows who have requested it have received the following issues of the *Journal of the American Rocket Society*:—Vol. 22, Nos. 5, 6; Vol. 23, Nos. 1 to 4.

The last of the thousand copies of the booklet *The Artificial Satellite* was sold during the year, and we cannot fill any further demands.

The number of copies of the *Journal* now printed has been increased to 4,000.

From this year, the printing of a separate *Annual Report* will cease, and this material will be combined with the November issue of the *Journal*. This

will have many advantages: not only will it result in a substantial reduction of our postage bill, but it will enable us to print a bigger *Journal*—one of at least sixty pages per issue.

During the year a controversy arose over the publication of a review in the *Journal* strongly criticizing a book by a Member of the Council as a result of which the practice of the Publications Committee on such matters was set out on p. 141 of the May *Journal*. All books are sent to the reviewers who, in the editor's opinion, are best qualified to deal with them. The opinions expressed are entirely the responsibility of the reviewer, and can be challenged by any author on matters of fact. We welcome forthright reviews, which are far more helpful to readers than non-committal ones, and every effort is made to ensure that they are authoritative.

I might also mention that for some time a rule has been in existence to the effect that no book by a Council member be reviewed by one of his colleagues!

Books, Radio, Television and Films

The most important astronautical film shown here during the year was *The War of the Worlds*, George Pal's brilliant interpretation of the Wells classic. Though it was certainly very bad propaganda for space-flight, the opening Bonestell sequences were of great beauty and interest, and perhaps foreshadowed what we may expect to see in the rumoured *Conquest of Space*.

Equally bad propaganda, though again very good entertainment, was the TV serial *The Quatermass Experiment*, which terrified a fair number of British viewers during the summer of 1953. We wish that producers would learn to use space-flight as something else besides a vehicle for horror. . . .

Your Chairman was interviewed on February 11, 1953, in the TV programme "Press Conference," his interrogators including the Editor of the *Observer*, the assistant editor of *Aeronautics*, and Dr. Alan Hunter of Greenwich.

The newspaper and magazine coverage of the subject is now far too extensive to summarize, and members are referred to our regular feature *From the World's Press*. I would, however, like to acknowledge again *Time* magazine's generosity in presenting Artzybasheff's original cover painting for their issue of December 8, 1952, illustrating the science feature "Journey into Space."

The number of books on space-flight and related subjects shows no sign of abating. One of the most important was the controversial *Across the Space Frontier*, based on the *Collier's* symposium. Several books by B.I.S. members were published, including Eric Burgess' *Rocket Propulsion*, V. A. Firsoff's *Our Neighbour Worlds*, G. V. E. Thompson's *The Adventure of Space Travel*, Gatland and Kunesch's *Space Travel* and P. A. Moore's *A Guide to the Moon*.

More are known to be on the way, and the situation is rapidly arising where the task of writing a new book on astronautics is as nothing to the problem of finding an original title for it!

Administration

One item of formal business arises here. It is necessary to move a special resolution to alter Article 30 of the Articles of Association, to read as follows:—

"Any group of members in one area may, subject to the sanction of the Council, form a Branch and elect a Branch Committee to act within the limits imposed by the Council."

The present articles require a fixed number of seven members, and the proposal is to leave the actual number in future to be fixed by the Council according to the facts of each particular case.

During the year, steady progress was made towards increasing the efficiency of the office and the general secretarial administration.

I would like to express my thanks to our Secretary, Mr. L. J. Carter, and to Mr. P. R. Freshwater, who has performed valuable work as Assistant to the Secretary, and also to Mr. C. R. Armstrong, who helped with the preparation of the List of Members. This proved to be a tremendous task, which can certainly only be undertaken at very infrequent intervals—at the most, every three years.

Hermann Oberth Medal

During the year the Society was honoured by the presentation to Mr. A. V. Cleaver of the Hermann Oberth Medal by the *Gesellschaft für Weltraumforschung*.

Subscriptions

I would now like to remind members of a very simple way in which they can assist both the Society and themselves. This is by signing the Banker's Order sent out with the next subscription reminder, so that in future their bank automatically pays their fees at the correct time. In this way they not only save themselves bother, but they relieve the office of a considerable amount of work, since there is then no need to send out receipts, as there is no possibility of remittances being lost in the post.

This is a simple service which everyone with a bank account can perform. There must also be many members who are in a position to compound their subscriptions, so that a lump sum payment makes them life members. The rates are as follows:—

Age Limits	Membership	Fellowship
	£	£
20–25	36	45
26–30	33	40
31–35	30	36
36–40	27	32
41–45	24	29
46–50	22	26
51–55	20	23
over 55	18	21

All moneys received from this source are placed in a special deposit account which, together with such other sums as can be spared from the general funds, enables the Society to build up a capital reserve. The amount now standing in this account is £1,100. The existence of such a fund is essential for the Society's continued progress, for in a relatively few years it may be necessary for us to move to larger premises—perhaps even to purchase some property—and this will not be possible unless we possess considerably more capital than we do to-day.

As a final comment on this subject, I would like to remind members of the possibility of making bequests to the Society. This is the way in which, during the past, most of the established scientific societies built up their capital.

Accounts

Once again I am pleased to report that our accounts disclosed a satisfactory position, owing largely to the very careful budgeting of our Secretary.

The expenditure on publications shows a considerable increase over last year, owing largely to the cost of the List of Members.

Once again we have been able to transfer the sum of £200 to Subscription Reserve Account, to cover the Society's expenses for the last quarter of the calendar year.

The amount standing to the credit of our Compounded Subscriptions Reserve Account has increased from £755 to £1,035. We are continuing to keep this in a separate fund to cover the eventual cost of the Society's own premises.

The Council

During the year the Council met 11 times. The amount of work to be dealt with remains at a very high level, and besides the routine running of the Society, the affairs of the I.A.F. now take up a considerable amount of time. The Membership Committee, which was set up last year, has functioned very well and has relieved the Council of much work.

Under Article 15 of the Articles of Association, four members of the current Council have retired, viz.: Messrs. E. Burgess, A. C. Clarke, A. E. Slater and R. A. Smith.

In view of the fact that I expect to spend most of 1954 in the United States, I have decided not to stand for re-election this year, but of course I shall continue to assist the Society in every way possible, both here and abroad.

I would like to conclude by once again expressing my gratitude to all those, at London and at the Branches, who have devoted so much of their time to the Society.

With this Report, I conclude my third consecutive year as Chairman of the Society, and now hand over to my successor. I have every confidence that he will be able to record continuing progress next year, and I would also like to express the hope that the Society is now settling down, as it were, after the perhaps too rapid expansion of the post-war years. We must not forget that we will have a very long period ahead of us when the achievement of space-flight will still seem far distant, and it may be necessary to maintain enthusiasm during times when there is little progress to report. That is why it is necessary to base the Society firmly on a secure scientific foundation, and above all not to "oversell" space-flight to the public. In a relatively short period of time, the B.I.S. has been able to establish itself as a serious and competent organization, regarded with respect in all parts of the world.

May it remain so in the future, so that when the history of astronautics comes to be written, we can be proud of the share our Society played in the final achievement.

ARTHUR C. CLARKE,

Chairman of the Council.

THE FOURTH INTERNATIONAL ASTRONAUTICAL CONGRESS

Zurich, 1953

The fourth international meeting of representatives from the astronautical societies which form the I.A.F. took place in Zurich during the week of August 2–8, 1953. On this occasion the Swiss Astronautical Society (S.A.A.) acted as host and all the official business and lectures were carried on in the Eidg. Technischen Hochschule—the very fine technical college of which Zurich is justifiably proud.

Ten member societies of the I.A.F. (Argentina, Austria, Denmark, Germany, Holland, Italy, Sweden, Switzerland, U.K. and U.S.A.) were represented and there were representatives and observers present from four other astronautical societies who were not members of the federation. The Norwegian Society did not send a delegate to the Congress, nor did the Spanish A.E.A. and several American societies. However, a new Spanish Society, which hopes to amalgamate with the older A.E.A., sent an observer.

The British delegation was fairly strong, including Messrs. A. C. Clarke and L. J. Carter (the official delegates), E. Burgess, G. Ramsden, A. C. Rotherham, W. Fielding and Doctors F. Singer, A. E. Slater and L. R. Shepherd (I.A.F. Vice-President).

The official proceedings of the Congress opened at 8 a.m. on Monday, August 3 at the Studentheim of the E.T.H., when the official delegates met to conduct the I.A.F. affairs, under the Chairmanship of the President, Dr. E. Sänger. This session began on a sad note, as the President made his unhappy announcement of the unfortunate accident which resulted in the death of the Hon. Vice-President, Dr. G. Loeser, on July 21, 1953. Dr. Loeser was killed when a helicopter, in which he was carrying out meteorological observations in Nebraska, crashed and burst into flames, with the loss of all who were on board. Mr. A. G. Haley, the senior Vice-President, gave a short appreciation of the work which Dr. Loeser had done on behalf of the I.A.F., after which the delegates, at the request of the President, rose in homage to their late colleague.

After the delegates had formally introduced themselves and given the current membership figures of their respective societies (see Appendix), the minutes of the plenary meetings of the third Congress (Stuttgart, 1952) were adopted and the meeting proceeded with the business of electing new societies to membership of the I.A.F. Three new societies (from South Africa, Yugoslavia and U.S.A.) had applied for membership and all were elected. The Yugoslav society did not send a representative but had authorized the I.A.F. secretary to vote on its behalf. Mr. A. C. Rotherham of the B.I.S. acted as delegate for the South African Interplanetary Society.

The business of the first day included a discussion of the motion to amend the I.A.F. constitution to permit a new class of membership, viz. "Corresponding Membership" to enable non-astronautical societies to be associated with the I.A.F. Although, in correspondence prior to the Congress, all but two of the member societies (the A.R.S. and B.I.S.) had expressed themselves in favour of this amendment, there was a tendency for many of the delegates to revise

their views, and after discussion it was decided that no decision should be taken on this matter during the Zurich Congress. The proposal will be re-considered at the 1954 Congress.

The remaining important business of the first day consisted of the report of the I.A.F. Secretary, Mr. Stemmer, including his budget proposals for the coming session; the election of committees (the Finance Committee; the committee for Classification and Abstracting; a Credentials Committee to consider the suitability of societies applying for membership) and a report, read by Mr. Haley of the work carried out during the year by Dr. Loeser in his endeavours to secure the admission of the I.A.F. to the International Council of Scientific Unions. The I.A.F. application to join the I.C.S.U. was later withdrawn after it was found that the federation did not conform to the requirement that it should have existed for at least six years.

The Finance Committee, consisting of Mr. A. Haley (Chairman), L. J. Carter, H. J. Kaeppeler, Ing. Ruckert and Professor T. Tabanera, met during the evening to consider the I.A.F. budget for 1953–54, and reported to the delegates during the plenary meeting on the following morning (August 4). The Finance Committee report included a recommendation for a sum of $250 to be set aside, out of the I.A.F. funds, to provide financial assistance to any smaller society organizing an I.A.F. Congress. The Austrian G.f.W. who were subsequently allocated the duty of organizing the next Congress, stated that they would not draw upon this fund.

A motion put forward by the B.I.S. to amend Article 28 of the constitution, according to which one of the Vice-Presidents must belong to the member society organizing the current Congress, was considered by the delegates, but was defeated. The B.I.S. view was that this placed an unnecessary restriction on the choice of candidates for the vice-presidencies and had, in fact, not been carried out in respect of the 1953 Congress. However, the majority of the delegates voted in favour of retaining the rule.

An important matter discussed during this session concerned the possibility of the I.A.F. publishing a scientific journal on astronautics, with the suggested title "Acta Astronautica." There was unanimous approval for this step forward, though there was considerable difference of opinion as to the manner in which this should be done and when it would be possible to start publication. Two main avenues were open to the I.A.F.; the first way would involve the I.A.F. in accepting complete responsibility for publication and merely paying a printer to produce and possibly distribute the journal; the second way would not involve the I.A.F. in any responsibility other than the provision and editing of the material for the journal, which would be published by some reputable publisher, who would undertake all the financial and administrative responsibilities. Professor Hecht of the Austrian G.f.W. had already been actively pursuing the second possibility, and mentioned several publishing houses who might be prepared to undertake the task. It was finally decided that a committee should be set up under the Chairmanship of Professor Hecht to consider the problems involved in this undertaking and report its findings back to the following Congress. The committee subsequently met on the morning of Saturday, August 8, and set up a smaller working party consisting of

Professor Hecht, Dr. Sänger, Dr. Casiraghi, Mr. Ordway and Dr. Shepherd, to work out some of the major technical problems associated with the publication.

Two final items were considered on the Tuesday morning, viz. the choice of venue and time of the next Congress, and the election of officials. It was suggested by the Austrian representatives that they should act as hosts next year and that the Congress should take place at Innsbruck during the first week in August, 1954. This proposal was accepted.

Only one nomination was received for the office of President of the I.A.F. This was the B.I.S. nomination of Mr. Fred Durant, President of the A.R.S. during the current year. Dr. Sänger indicated that he would not offer himself for re-election and in the ballot which followed Mr. Durant was elected unanimously. The first nomination for Vice-President was Professor Hecht, the chief delegate of the Host Society for the 1954 Congress, and, in accordance with Article 28 of the constitution this election was automatic. Mr. Haley was nominated once again for Vice-President and was elected unanimously. Mr. Stemmer was also re-elected Secretary.

One remaining item of business was discussed during a short session held on Saturday morning, August 8. This concerned the future activity of the I.A.F. A number of delegates felt the need for extending the activity of the I.A.F. beyond its present limits. However, only one suggestion was made, this by the Italian representative Dr. Casiraghi, viz., that the I.A.F. should encourage individual work in astronautics by offering periodically a medal for the best paper written on some technical aspect of the subject. The suggestion was adopted unanimously and it was decided to call the award the "Gunter Loeser Memorial Medal." It was decided furthermore that the subject for the first award should be "The Economics of Space-Flight" and that the entries should be submitted before January 1, 1955.

The part of the Congress devoted to the technical lectures began on Tuesday afternoon in the physics department, with an opening address by Professor J. Ackeret. This took place in what must be one of the largest and most magnificent lecture theatres in the world. The front of the theatre was provided with such a multiplicity of switches and gadgets that it was, perhaps, not surprising that during the first lecture by Dr. Irene Sänger-Bredt, a state of near-chaos prevailed as each lantern slide was shown. The remainder of the lecture programme was held in the more modest surroundings of the mechanical engineering department.

It is not proposed to give details of the lectures here as they are, in fact, to be published by the S.A.A. in a large bound volume. The B.I.S. papers were contributed by Messrs. Gatland, Dixon and Kunesch ("Fabrication of the Orbital Vehicle"); Dr. F. Singer ("A Minimum Orbiting Instrument-Carrying Rocket") and Mr. W. N. Neat ("Limitations in the Performance of Chemical Rockets in Astronautics").

A film show made up of many well-known rocket films, including the epic voyage of the astronautical mice, was shown on Thursday evening in the physics lecture theatre.

The social side of the Congress was not lacking. A reunion of the delegates took place on the Sunday evening preceding the Conference, at the social

meeting place and restaurant "Zum Rüden." Here many delegates came together once again to renew aquaintance and to find new friendships. On Wednesday afternoon a cruise across the Zürichsee, arranged by the Swiss Aero Club, took place in bright sunshine and helped to restore the flagging energies of the delegates. Another item on the social programme, provided this time for the ladies, was a visit to the chocolate factory of Lindt and Sprüngli at Kilchberg.

The grand finale of the Congress was the reception on Saturday evening, when a dinner was provided for all the delegates at the Zum Rüden. This was attended by the mayor and other notabilities and many humorous speeches, including a trilingual address from Dr. von Karman, were given. At this gathering the new President, Mr. Durant, on behalf of the delegates, presented an inscribed silver cigarette case to the retiring President, Dr. Sänger.

LIST OF I.A.F. MEMBER-SOCIETIES

			Membership at Stuttgart Congress
Argentina.	Sociedad Argentina Interplanetaria (S.A.I.)	185	*85*
Austria.	Oesterreichische Gesellschaft für Weltraumforschung (O.G.f.W.)	122	*102*
Denmark.	Dansk Selskab for Rumfarts-Forskning (D.S.f.R.)	45	*32*
Germany.	Gesellschaft für Weltraumforschung (G.f.W.)	714	*528*
Holland.	Nederlandse Vereniging voor Ruimtevaart (N.V.v.R.)	160	*120*
Italy.	Associazzioni Italiana Razzi (A.I.R.) ..	105	*88*
Norway.	Norsk Interplanetarisk Selskap (N.I.S.) ..	*(Not represented)*	*28*
Sweden.	Svenska Interplanetarisk Selskap (S.I.S.)	42	*26*
Switzerland.	Schweizerische Astronautische Arbeitsgemeinschaft (S.A.A.)	62	*49*
Union of South Africa.	South African Interplanetary Society (S.A.A.)	44	—
U.K.	British Interplanetary Society (B.I.S.) ..	2,250	*1,970*
U.S.A.	American Rocket Society (A.R.S.) ..	2,301	*2,011*
	Chicago Rocket Society (C.R.S.)	*(Not represented)*	*(Not known)*
	Detroit Rocket Society (D.R.S.)	"	*85*
	Pacific Rocket Society (P.R.S.)	"	*295*
	Philadelphia Astronautical Society ..	38	—
Yugoslavia.	Yugoslav Astronautical Society	65	—
	Total	*6,133*	*5,419*

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Donations totalling £296 5s. 9d. were received from 481 members during the year and are gratefully acknowledged.

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The cost is 10/- per strip, post free, which works out at the modest sum of 2½d. a picture. Orders should be sent to the Secretary at 12, Bessborough Gardens, S.W.1.

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of the

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22, Gladstone Place, Brighton, Sussex.

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895. WALKER, Ian Frederick Macey, *B.Sc., Grad.I.E.E.* — FELLOW
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813. WALKER, Lawrence Hosier — FELLOW
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871. WALKER, Ronald — MEMBER
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614. WALKER, Ronald Selby, *Comp.R.Ae.S.* — MEMBER
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2770. WALLACE, Charles James — MEMBER
37, Kensington Mansions, Trebouir Road, London, S.W.5.

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442. WALLACE, Richard Bruce — MEMBER
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406. WALMSLEY, John Henry, *Grad.I.Mech.E., A.F.R.Ae.S.* — FELLOW
80, Pitts Lane, Woodley, Berks.

349. WALPORT, Samuel, *M.R.C.S., L.R.C.P.* — FELLOW
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1703. WALSH, John Denis, *A.F.R.Ae.S.* — FELLOW
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980. WATERS, Paul — MEMBER
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2771. WATNEY, John Basil — MEMBER
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2886. WATSON, William Kemper — MEMBER
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1630. WAY, Allan Jay, *D.Sc., LL.D., Ph.D.* — FELLOW
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2456. WEBB, Alan Arthur — MEMBER
40, Lyncroft Avenue, Pinner, Middx.

2887. WEBB, Horace Gilbert William — MEMBER
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1079. WEBB, Reginald Ernest — MEMBER
10, The Crescent, Pewsey, Wilts.

121. WEBB, Ronald E. *B.Sc., A.C.G.I.* — FELLOW
10, Mount Harry Road, Sevenooks, Kent.

3259. WEBDALE, Gerald Thomas — MEMBER
9, North Field Avenue, Wells, Norfolk.

287. WEBSTER, Douglas — MEMBER
"Idlewild," Fountainhall Road, Aberdeen, Scotland.

713. WEEDALL, Percy Norman — MEMBER
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1497. WEEKS, Gerald Christopher — MEMBER
23, Welford Road, Sutton Coldfield, Warwickshire.

2022. WEETCH, John — MEMBER
5, Allcroft Road, Kentish Town, London, N.W.5.

2073. WEIBEL, John Lawrence, *B.S., D.V.M.* — FELLOW
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3190. WEINER, Richard Stanley — MEMBER
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2962. WEISS, Maurice — MEMBER
1563, Inwood Avenue, New York 52, N.Y., U.S.A.

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2394. WENDT, Gerald, *Ph.D.* — MEMBER
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3191. WERNER, Ian Robert — MEMBER
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2687. WEST, Donald Sydney — MEMBER
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2516, Harriet Avenue South, Minneapolis 5, Minn., U.S.A.

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640. WESTFIELD, Alec — MEMBER
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698. WHATMORE, Bernard Frank — MEMBER
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3192. WHEATLEY, John Brooks, *A.B.* — FELLOW
5019, N. Meridian Street, Indianapolis 8, Ia., U.S.A.

2222. WHEELER, John Fores — MEMBER
2644, N.E. Bryce Street, Portland 12, Oreg., U.S.A.

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3, Tyrrell Avenue, Bexleyheath, Kent.

1632. WHITCUTT, Roger — MEMBER
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1983. WHITE, Ambrose Joel — MEMBER
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2825. WHITE, Ewart Charles — FELLOW
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2964. WHITE, Gordon Joseph — FELLOW
1262, Greenford Road, Greenford, Middx.

1678. WHITE, James — MEMBER
29, Colinpark Street, Springfield Road, Belfast, N. Ireland.

2145. WHITE, Jerry Gillman — MEMBER
215, Essex Square, West Harnham, Salisbury, Wilts.

2531. WHITE, Raymond William — MEMBER
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758. WHITE, Reginald Graham, *M.B.*, *B.S.*(*Lond.*), *M.R.C.S.*(*Eng.*), *L.R.C.P.*(*Lond.*) FELLOW
P.O. Box 84, Rusapi, S. Rhodesia, E. Africa.

2965. WHITE, Jr., Thomas Otis MEMBER
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1426. WHITEHEAD, Samuel Derrick MEMBER
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2223. WHITWORTH, James Peter, *Grad.R.Ae.S.* FELLOW
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151. WHOLEY, Max Wingfield, *A.M.I.Mech.E.* FELLOW
10/492, Allengauj, Kanpur, U.P., India.

999. WICKENS, Alan Herbert, *B.Sc.*(*Eng.*), *D.L.C.*, *Grad.R.Ae.S.* MEMBER
10, The Quadrant, Warwick Road, Coventry, Warwicks.

696. WICKS, Harold David MEMBER
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3100. WICKWIRE, Austin, *Mech.E.*, *M.I.Ae.S.* MEMBER
308, West Paseo de Cristobal, P.O. Box 644, San Clemente, Cal., U.S.A.

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1149. WILD, Anthony MEMBER
767, High Road, Leytonstone, London, E.11.

3261. WILDER, John Patrick Charles MEMBER
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1558. WILKINS, Donald Spalding, *B.Sc.* FELLOW
20, Holly Road, Edgbaston, Birmingham, 16.

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35, Fairlawn Avenue, Bexleyheath, Kent.

769. WILKINS, James Humphrey, *Grad.I.E.E.* FELLOW
46, Beacon Road, Loughborough, Leicstershire.

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3, Birkendale, Sheffield 8, Yorks.

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28, Recreation Road, Tilehurst, Reading, Berks.

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2041, Dryden Road, Houston 25, Texas, U.S.A.

1427. WILLIAMS, David Michael Hessom MEMBER
89, Barton Road, Scotforth, Lancaster, Lancs.

3262. WILLIAMS, David Victor James MEMBER
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1039. WILLIAMS, Geoffrey Edward, *A.F.R.Ae.S., Grad.I.Mech.E.* FELLOW
9, Elmwood Crescent, Luton, Beds.

730. WILLIAMS, John Ames, *B.Sc., A.M.I.Mech.E.* FELLOW
11, Moreton End Lane, Harpenden, Herts.

2225. WILLIAMS, John Colin MEMBER
203, Albany Street, London, N.W.1.

3102. WILLIAMS, Raymond MEMBER
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567. WILLIAMS, Ronald Sidney, *B.Sc.* MEMBER
32, Ashbourne Road, Mitcham, Surrey.

1655. WILLIAMS, Terence Anthony MEMBER
47, Aubert Court, Highbury, London, N.5.

1060. WILLINSKI, Martin Irwin, *B.S.* MEMBER
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1908. WILLIS, Charles Milford MEMBER
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2759. WILLITS, Frederick Thomas MEMBER
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1370. WILLOUGHBY, Geoffrey Charles Alfred MEMBER
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376, Moseley Road, Balsall Heath, Birmingham 12.

2888. WILSON, Mrs. Caroline Elsie MEMBER
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2458. WILSON, Donald MEMBER
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1371. WILSON, Rev. Ernest Stanley MEMBER
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1063. WILSON, Geoffrey Leonard, *M.A., B.Sc., A.M.I.E.E.* MEMBER
3, Sheridan Road, Merton Park, London, S.W.19.

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3015. WILSON, Hugh Geoffrey Birch, *M.A., M.P.* MEMBER
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2889. WILSON, James Edmund MEMBER
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43. WILSON, Lawrence, *M.A., LL.B.* FELLOW
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2460. WILSON, Michael Edwin MEMBER
2, Skipton Road, Crosshills, nr. Keighley, Yorks.

2296. WILSON, Michael John MEMBER
c/o A. C. Clarke, 88, Nightingale Road, London, N.22.

2297. WILSON, Peter Anthony Gordon, *B.Sc.* FELLOW
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1656. WILSON, Philip Field MEMBER
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2146. WILSON, Roy Allan, *B.Sc.* FELLOW
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1817, Schiller, Little Rock, Ark., U.S.A.

1559. WILSON, William Gilbert Steward, *B.Sc.* FELLOW
2, Wellfield Road, Blackburn, Lancs.

1372. WILTSHIRE, Leonard Charles, *M.Sc., A.Inst.P.* FELLOW
15, Frankland Road, Chingford, London, E.4.

314. WINDSOR, Maurice William, *B.A.* MEMBER
193, Cherryhinton Road, Cambridge.

1373. WINGFIELD-DIGBY, Frederick George MEMBER
33, Harrington Gardens, London, S.W.7.

162. WINICK, Miss Helen MEMBER
c/o 12, Budleigh Crescent, Welling, Kent.

2690. WINNING, Jr., Herbert, *B.Sc., A.M.I.Chem.E., A.R.I.C.* FELLOW
99, Clarence Drive, Glasgow, W.2.

2298. WINTERBOURNE, Jack Leslie MEMBER
10, Grove Crescent, Kingston-on-Thames, Surrey.

1500. WINTLE, William Thomas MEMBER
16, Murray Road, Southville, Bristol 3, Glos.

2074. WINTRINGHAM, Oliver Joseph, *B.A.* MEMBER
29, Hawthorn Road, Wallington, Surrey.

1577. WISEMAN, Thomas Settimo MEMBER
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1305. WITHAM, Robert George MEMBER
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3263. WITHERS, Ralph MEMBER
20, Pooksley Road, Orpington, Kent.

1633. WITHINGTON, Richard Henry, *A.I.Mar.E.* MEMBER
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2033. WOLTON, William Philip MEMBER
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2396. WOLVERSON, Tom FELLOW
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1560. WOOD, Alan Spencer, *B.Sc.* FELLOW
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2966. WOOD, Archibald Cuthill MEMBER
17, Woodhouse Close, Perivale, Middx.

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2498. WOOD, Henry Sinclair, *D.Arch.* (*Edin.*), *Dip. T. & C.D.* (*Edin.*), *A.R.I.B.A., A.R.I.A.S., A.M.T.P.I.* FELLOW
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2532. WOOD, John Neville MEMBER
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1801. WOOD, Leslie Arthur MEMBER
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2533. WOODFORD, Lt. John Aubrey, *B.Ph., B.A.* MEMBER
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867. WOODROW, Mrs. Kathleen Mary MEMBER
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857. WOODROW, Reginald James, *B.Sc., A.M.I.Mech.E.* MEMBER
3, Windermere Crescent, South Shields, Co. Durham.

571. WOODRUFF, Capt. Alan Ivor MEMBER
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2581. WOODS, Brian MEMBER
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1984. WOODS, Robert Octavius MEMBER
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313. WOODTHORPE, John Malpas — MEMBER
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1706. WOODWARD, Bruce Allyn — MEMBER
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2525. WOODWARD, Miss Phyllis Mary — MEMBER
St. Joseph's, Upton-on-Severn, Worcester.

2582. WORKMAN, Geoffrey Herbert — MEMBER
90, Edward Street, Oldham, Lancs.

2026. WORT, Dennis John Howard — MEMBER
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3283. WRIGHT, Geoffrey, *B.Sc.* — MEMBER
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673. WRIGHT, Harry Anthony Grosvenor — MEMBER
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1501. WRIGHT, Henry John, *A.M.I.E.E., M.Inst.F.* — FELLOW
c/o 10, Slamannan Road, Falkirk, Stirlingshire.

792. WRIGHT, Ian Brodie — MEMBER
1, Mirfield Drive, Monton, Eccles, nr. Manchester.

27. WRIGHT, Michael Stephen, *B.Eng.* — FELLOW
39, Watson Road, Rotherham, Yorks.

1912. WRIGHT, Robert George — MEMBER
5, Cantsfield Street, Edge Hill, Liverpool 7.

1744. WRIGHT, Robin Dakin, *B.Sc., A.M.I.E.E.* — MEMBER
Sunnyhill, Lodge Road, Caerleon, Mon.

3194. WUNDERLICH, Stanley, *B.E.E.* — MEMBER
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X

2075. XENAKIS, George — MEMBER
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Y

1072. YARNOLD, Alan Denis — MEMBER
2, Marion Court, Tooting High Street, London, S.W.17.

2761. YATES, Frank Leo — MEMBER
7, Ferndale Avenue, Frankby, Wirrall, Cheshire.

3121. YEANDEL, Jonathan David, *Grad.I.P.E.* — FELLOW
54, Springfield Avenue, Bishopston, Bristol 7, Glos.

856. YORKE, Major Frederick Augustus, *M.C.* — MEMBER
97, Overstand Mansions, Battersea Park, London, S.W.11.

2762. YOUNG, Alan Douglas — MEMBER
786, Hertford Road, Freezywater, Enfield, Middx.

2763. YOUNG, Brian William — MEMBER
5, Perryhow, Worcester Park, Surrey.

1930. YOUNG, Edward John — MEMBER
Box 48, Tuxedo Park, New York, U.S.A.

1707. YOUNG, John Cronin — MEMBER
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1913. YOUNG, Ralph Harry Wilfred — MEMBER
107, Strathmore Road North, Gosforth, Newcastle upon Tyne 3, Northumberland.

2692. YOUNG, Robert William — MEMBER
7, Kellie Place, Alloa, Clackmannanshire.

2827. YOUNG, Ronald Victor — MEMBER
19, Nelson Crescent, Ramsgate, Kent.

221. YOUNG, Thomas Gregson, *A.M.Inst.B.E.* — MEMBER
Stathmos, Whalley Road, Great Harwood, Blackburn, Lancs.

Z

2583. ZAHAUI, YACOU — MEMBER
c/o C. W. Cain, 27, Chenistow Gardens, Kensington, London, W.8.

2828. ZETTLER, Raymond Paul — MEMBER
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2147. ZIEBLAND, Hans, *Dipl.Eng., A.M.I.Mech.E.* — FELLOW
92, Monkswood Avenue, Waltham Abbey, Essex.

1563. ZIMMERMAN, Charles — MEMBER
7, Stanley Gardens, Kensington, London, W.11.

2967. ZIMMERMAN, Charles Horton, *B.S.* — FELLOW
95, Alleghany Road, Hampton, Va., U.S.A.

2499. ZULIKOWSKI, F/Lt., Jozef Zbigniew, *D.F.C.* — MEMBER
R.A.F., Calshot, Fawley, Southampton.

542. ZUMPE, Herman Frank, *Dip.Ing.* — FELLOW
R.A.E., Westcott, nr. Aylesbury, Bucks.

850. ZWICKY, Professor Fritz, *Ph.D.* — FELLOW
California Institute of Technology, Pasadena 4, Cal., U.S.A.

MEMBERSHIP STATISTICS

The total number of members shown in this List of Members at 30th June was 2,263, consisting of 6 Honorary Fellows, 546 Fellows, and 1,711 Members.

The age distribution of members was as follows:—

Age 17 and under	6%
" 18/22	18%
" 23/27	26%
" 28/32	21%
" 33/37	11%
" 38/42	6%
" 43/47	4%
" 48/52	4%
" 53 and over	4%
	100%

The geographical distribution of members is as follows:—

British Isles	76%
U.S.A.	14%
Rest of the World	10%
	100%

A broad classification into occupations is as follows:—

1. Engineers (Qualified and Senior Men)	19%
2. Scientists and Research Workers	8%
3. Technicians	22%
4. Students (7% specialising in technical subjects)	15%
5. Others (including 10% Professional Men and 9% in the Armed Forces)	36%
	100%

An analysis of the membership resident in England into the various counties is as follows:—

Bedfordshire	28
Berkshire	24
Buckinghamshire	34
Cheshire	38
Devonshire	26
Essex	71
Gloucestershire	46
Hampshire	77
Hertfordshire	45
Kent	75
Lancashire	102
Leicestershire	23
London	327
Middlesex	104
Staffordshire	36
Surrey	111
Sussex	43
Warwickshire	85
Yorkshire	92
Others (counties with under 20 members in each ..	330
	1717

An analysis of the members according to the number of years for which they have been members is as follows:—

17%	have been members for 5 years and longer.
13%	" " " " 4 "
20%	" " " " 3 "
30%	" " " " 2 "
20%	" " " " 1 year
100%	

ACKNOWLEDGMENT

Particular thanks are due to Mr. C. R. Armstrong for his untiring assistance in the compilation of this List of Members.

THE BRITISH INTERPLANETARY SOCIETY

Secretarial address: 12 Bessborough Gardens, London, S.W.1.

Tel. TATe Gallery 9371.

Suggestions for Lecture Notes for B.I.S. outside speakers

General Advice

If a lecture based on *all* the following notes was given, it would last for perhaps a couple of hours. This would be most undesirable; the close attention of an audience cannot be held, by any speaker on any subject, for much over one hour—except under the most unusual circumstances. The first point to remember, then, is to time your lecture to last between 45 and 75 minutes.

This means, therefore, that lecturers are intended to use these notes as follows: try to include at least a passing reference to all the sections of the subject covered by the notes, but concentrate on those sections best suited to your own abilities and the known interests of your audience (e.g., astronomical, engineering, technical or non-technical, etc.).

Another strong argument in favour of keeping the main lecture not over-long is the need to reserve time for questions. There are almost invariably plenty of these—submitted, nowadays, in a friendly spirit of genuine information-seeking. One is, to-day, more often questioned about the need, value or purpose of interplanetary flight than its technical practicability. In answering any questions from technical specialists, if the individual lecturer does not personally know the full answer, it is better to admit this frankly, making it clear that the point in question has been considered by specialists in the B.I.S., and offering to obtain a written answer if required.

On this same matter of creating a generally favourable impression, it is preferable to deliver both the lecture and answers to questions in a matter-of-fact and light manner (but, of course, avoiding downright facetiousness). Show, in other words, that you can appreciate the fact that many astronautical ideas seem fantastic to laymen meeting them for the first time.

Include, of course, references to the activities of the B.I.S., where relevant, but avoid too much of the "sales-talk" atmosphere.

I. Introduction

The whole field of rockets and space travel is a very large one to try to cover in one lecture. There are many other applications of rocket propulsion besides the interplanetary one—military missiles (defensive and offensive), high-altitude rockets for meteorological and cosmic-ray, etc., research, aircraft propulsion and assisted take-off, etc. Many already of great practical importance.

Interplanetary idea, however, is an old dream of mankind, constantly encountered in mythology and fiction—Verne, Wells, etc. Rocket at last promises a means of making this dream come true.

Reasons:—(*a*) Rocket works best (highest thrust) in vacuum of space, beyond atmosphere.

(*b*) Rockets can deliver enormous power output for very small size and weight.

Both are very essential requirements for interplanetary flight.

II. Technical. How a Rocket Works

Airscrew generates thrust by the reaction from throwing backwards a mass of air (slipstream). Turbojet the same, except that it accelerates this air by passing it through a thermal cycle, instead of by an external mechanical device.

Like these, rocket motor is a "reaction-propulsion" system, but, *unlike them*, it carries within itself *all* the material required to make its propulsive jet, and does not rely on the atmosphere for any of it.

Emphasize—rocket works by pure reaction, does not "push" against the air outside. Thrust is the result of all the high pressures inside the motor pushing against its forward walls—at rear, no "wall," only open nozzle, so there is always a net forward force. (For popular talks, use analogies of gun recoil, balloon blown along by air rushing out of open neck, etc.)

Thrust is greater in vacuum because of more favourable expansion ratio; reduction of atmospheric back-pressure on nozzle ceases to hinder high-velocity exit of exhaust gases.

Thrust = Rate of change of momentum of exhaust gases.
= Mass flow x exhaust velocity.

E.V. is very fundamental parameter: specific impulse $= \frac{\text{E.V.}}{g}$

Specific impulse = thrust per unit wt. ejected per sec., simply
= reciprocal of specific fuel consumption. Latter is always high for rocket, cf. other engines, because rocket needs to carry its own source of oxygen as well as fuel.

BUT:—

Rocket, due to simplicity and high working pressures, is always itself small and light relative to its thrust output.

Explain, where desirable:

(*a*) Propellants = Fuel and Oxidiser.

(*b*) Fuel = Source of energy (e.g., petrol, alcohol, etc.).

(*c*) Oxidiser = Source of oxygen with which to burn fuel and release its energy (e.g., liquid oxygen, hydrogen peroxide, nitric acid).

Solid propellant rockets combine both fuel and oxidiser in one substance, e.g., gunpowder or cordite. —so do liquid "mono-propellant" rockets (e.g., nitromethane).

Actually, any chemical reaction releasing energy can be basis of a rocket system—for example, hydrogen and fluorine, or the decomposition of hydrogen peroxide by itself. Oxygen need not be present, though it usually is, i.e., most rockets work on a simple combustion reaction.

Combustion chamber temperatures = 2000 degrees C. or more.
" " pressures = 300 lb./sq. in. or more.

Very severe cooling problem. (Regenerative cooling, by one of propellants.)

Supersonic exhaust velocity, hence convergent-divergent nozzle (typical present E.V. = 6500 ft./sec.).

Propellant feed to chamber by:

(*a*) compressed air or nitrogen or high-pressure gas from burning charge —suitable for short burning times; or

(*b*) mechanical, often turbine-driven, pumps—for longer burning times.

Other engineering problems:—control valves (gas and liquid), propellant injectors for good mixing and atomization, etc.

Liquid-propellant rockets more important because:

(*a*) Thrust is controllable, on throttle lever as for other engines.

(*b*) Weight is much lower, for long burning times.

(*c*) More powerful propellants are available as liquids.

Velocity attained by a rocket in vacuo (air drag and gravity reduce this value) is given by:

$$\text{Rocket velocity} = \text{Exhaust velocity} \times \text{Log. (base e) Mass ratio.}$$

This is the fundamental rocket relation, obtainable from consideration of conservation of momentum between rocket and its exhaust.

$$\text{Mass ratio is:—} \frac{\text{Mass of rocket full of propellants}}{\text{Mass of rocket empty of propellants}}$$

Limit to mass ratio is set by structural engineering, but we can cheat this by building "step-rockets." Big rocket lifts smaller one, drops off when its propellants are exhausted, then smaller one begins to fire—it therefore adds its own velocity on to that already imparted to it by the earlier "step." In this way, the deadweight of structure, tankage, motors, etc., which has to be lifted at any instant is kept to a minimum.

e.g., WAC Corporal + V.2 (250 miles in 1949)

Step rockets are of great future importance.

Rocket projectiles could be landed by parachute, or by wings as gliders, in which latter case their range is extended.

Suggested references:

Sutton, "Rocket Propulsion Elements."
Clarke, "Interplanetary Flight."

III. Historical

Rocket first invented (probably by accident) by Chinese, centuries ago. Used as toy and crude weapon. Latter use spread through Asia to India, where British adopted it. Congreve's war rockets of Napoleonic Wars. Advent of improved guns relegated rocket to being simple firework, life-saving and signalling device, until turn of century, when it began to be studied scientifically by men who appreciated its potentialities for space-flight (Ziolkovski, Esnault-Pelterie, Goddard, Oberth).

Amateur rocket societies demonstrated practically most of the basic ideas in years between wars—circa 1930 (A.R.S. and, most of all, the German V.f.R.)

During war, Allies concentrated on simple, small, solid rockets, but Germans went ahead with the vastly more difficult, and technically much more significant, task of perfecting the liquid-propellant rocket.

Describe German war rockets, e.g.:—

(*a*) A.4 = V.2 (and ambitious projected developments, A.9 + A.10).

(*b*) Rheinbote (step-rocket).

(*c*) Defensive rockets (emphasize this: after V.2, many people think the rocket is purely a terror weapon) such as Wasserfall and Rheintochter.

(*d*) Other missiles, e.g., the anti-shipping Hs.293.

(*e*) Aircraft assisted take-off units of the Walter firm. (Here the rocket's ability to give high thrusts for low weight was the quality being exploited. The Allies also used R.A.T.O. but only solid units on small naval (carrier) aircraft.)

(*f*) Aircraft propulsion—e.g., Me 163 with Walter motor, climbed to 40,000 ft. in 4 mins. Terrific performance, due to rocket's high thrust at high altitudes, but only for a short time, due to rocket's heavy fuel consumption.

Since war, Americans have done most work, largely for economic reasons.

Describe:—

Bell X-1 supersonic research plane (pure rocket propulsion).

Douglas Skyrocket supersonic research plane (over 1200 m.p.h.).

Republic jet + rocket fighter (XF.91).

WAC Corporal, Aerobee, Viking, Lark, Firebird, GAPA, Convair, etc., missiles, for military or research purposes.

Vertical firings of V.2's, etc.

Officially announced project for an "Earth-satellite vehicle."

Project RAND publications on meteor danger, upper atmosphere research, etc.

Douglas X.3 and Bell X.2 (few details yet available).

We believe that the next few decades will show increasing use and development of the rocket for all the purposes covered by the above. Then, we shall see piloted rocket craft—"semi-aircraft" like the A.9 + A.10 or the Sanger project for a long-range rocket bomber, taking off vertically as projectiles, shooting beyond the atmosphere and falling back into it to glide at high speeds for long distances.

The orbital rockets (Earth-satellite vehicles) will follow—just small unmanned rockets with automatic instruments at first.

Finally, there will come the real crew-carrying spaceships. The whole thing will be a gradual development, and may take all of 50 years—but if present trends are maintained, there is little room for doubt about the final result.

Suggested references:

Ley, "Rockets, Missiles and Space-travel."

Gatland, "Evolution of the Guided Missile."

Clarke, "Interplanetary Flight," "The Exploration of Space."

IV. Spaceflight

Begin by describing Earth's gravitational field, orbital and escape velocities.

Gravity follows inverse-square law, so any point in space has its certain value of gravitational force. Emphasize for lay audiences: pure nonsense to talk about getting "beyond the pull of gravity"—gravity extends, decreasing all the way, right to infinity. We can, however, make sure that gravity will never recapture a rocket by giving it an appropriate velocity.

If, e.g., a rocket climbed up from the Earth into a circular orbit around it, having (when it got there) a certain velocity, then a centrifugal force, depending on that velocity, would act on the rocket. If we made the velocity right, then C.F. would just balance gravity. MOREOVER, the rocket would retain the velocity indefinitely, even with its motors shut down, there being no air-resistance to slow it down. This is theory of the space-station, orbital rocket or earth-satellite vehicle. Cf. Moon as satellite of Earth, or Earth of Sun—or stone whirling round on a piece of string.

For a given point in space, there is another interesting critical velocity, called "escape velocity." This is the velocity, which, if a body possesses it, enables that body to continue on its way, all the way to infinity—always losing velocity under the (continuously diminishing) influence of gravity, but being brought to rest only at infinity.

Orbital and escape velocity are terms long used by astronomers—not recent inventions of the interplanetary enthusiasts.

Spaceship would therefore use its motors to accelerate up to escape velocity (25,000 m.p.h.) and would then shut them down. Acceleration would last only a short while—say 10 mins—and accelerations employed would be quite tolerable to the crew—say 2 *g* effective (= 3 *g* true). Ship would never be travelling fast enough inside the atmosphere to heat up appreciably (there is practically no air beyond 100 miles or so).

While coasting (i.e., for most of voyage), the motors would only occasionally be re-started, to make any necessary course corrections. Rocket steering jets might also be used for this purpose.

During this coasting period, the ship and all its contents, including the crew, would be equally under the influence of retarding gravity—these conditions being described as "free-fall"—so that the crew would experience apparent weightlessness and have no sense of "up" or "down." Would be like passengers in falling lift cage, the physiological effects of this are not known: it would not hinder breathing, digestion or excretion (all muscular functions, independent of *g*) but might affect balance and have bad psychological effects. We shall know when first piloted rocket returns from a flight outside the atmosphere. Anyhow, solution known—spin the ship about its axis and use C.F. as substitute for gravity.

(N.B.—This "weightless" effect is nothing whatsoever to do with the gravitational null point, e.g., between Earth and Moon, or with the decreased pull of gravity. It would happen only few miles from Earth if motors were cut off.)

As the ship passes the null point, it begins to accelerate towards the destination planet, and soon must be turned round by the steering jets and the main motors started again to brake for a slow landing—"sitting on the jet." If the destination planet has an atmosphere, then some air-braking may be possible, saving fuel.

Of course, enough fuel must be carried to permit a return take-off and flight back to Earth. (Lecturers may be surprised to discover how many people do not realize we envisage doing this!)

Power requirements for interplanetary flight are so vast that chemical fuels may never be adequate. Step rocket principle will certainly be employed, and use probably made of "orbital refuelling"—i.e., having the spaceship wait in a stable orbit above the Earth while other rockets refuelled it, before it continued its journey. Even so, application of nuclear energy to rocket propulsion will probably have to come before full realization of spaceflight. Probably by using a nuclear reactor (atomic pile) to heat up a mass of some light gas, such as hydrogen, as material for a rocket jet.

Meteorites—appreciable danger, but not serious one. We know a lot about meteor frequency and can calculated probability of hit—not high. Even then, few hits serious.

Cosmic rays—intensity does not rise continually with height; we believe from high-altitude rocket measurements that there is no great danger, but further information needed.

Cabin air-conditioning—not much worse problem for spaceship than for submarine or stratosphere aircraft.

Communication—short-wave radio O.K. Present equipment can reach planets. (N.B., also radar altimeters, etc.)

Navigation—can always see the stars!

All these problems are important, but it is the propulsion one which is fundamental—and rocket principle is answer to that.

Suggested references:

Ley, "Rockets, Missiles and Space-travel"; "The Conquest of Space."
Clarke, "Interplanetary Flight"; "Exploration of Space."
Gatland, "The Evolution of the Guided Missile."

V. Astronomical

No need to develop this section fully as excellent books are available (see References at end).

Places we could go:—

Moon—first objective—	¼	million	miles	away	
Venus (nearest)	26	,,	,,	,,	} Describe Solar system briefly.
Mars	35	,,	,,	,,	
Pluto	3500	,,	,,	,,	
Sun	93	,,	,,	,,	
Nearest star	25×10^{12}	,,	,,	,,	

Our Sun is just one of millions in Milky Way galaxy—and there are millions of other galaxies.

(It is now believed that many other stars have planets, but travel to them belongs to remote future—they are so far away.)

If intelligent life exists elsewhere in our own Solar System, it is probably very different from Man—but still possible.

Planetary environments hostile to Man—e.g., methane atm. at high pressure and 3 *g* on Jupiter; even on Mars, it is unlikely we could survive without space-suits, and Moon is of course airless. Big job of colonization (civil engineering) to be done—worth mentioning we shall never be *quite* sure what planetary surface conditions are like until we go there.

Journey times (maximum—could be shorter if we could spare the energy):
Moon—100 hours. Mars—237 days. Venus—146 days.

Suggested references:

Whipple, "Earth, Moon and Planets."
Spencer Jones, "Life on Other Worlds."
Ley and Bonestell, "Conquest of Space."
Davidson, "Astronomy for Everyman."

VI. Philosophical

Especially if all the planets are so inhospitable, why do we want to cross space?

One can supply all sorts of rationalized (though quite valid) answers to this question, which may satisfy the "practical" man, e.g.:—

"Think of the new knowledge!" (H. G. Wells, in "First Men in the Moon").
And knowledge usually pays large, unforeseen dividends. But also worthwhile for own sake.—Gain to astronomy (immense), physics, etc.
Value of space-station as radio-relay, for weather forecasting, etc.

One can also fall back on the argument that, after all, Columbus hardly foresaw all the consequences of discovering America when he set sail.

BUT:—Honest answer is probably that we do these things because we *must* —for the same "reasons" that have motivated all explorers, all discoverers, throughout history.

Effects of spaceflight may be greatest in indirect ways—cf. theory of evolution, Copernican astronomy, Freudian psychology, etc.—i.e., in changing our perspective on the universe and our whole outlook. At least the tribal squabbles which have wasted so much of the time and treasure of our species in the past will appear in their true petty scale, seen against the backcloth of the Universe.

Suggested reference:

Clarke, "Exploration of Space" (Chapter 18).

INTERPLANETARY FILM STRIP

The Society has now prepared, for the benefit of lecturers, a 35 mm. film strip containing 50 pictures showing modern rockets, orbital satellites, spacesuits, spaceship designs, and astronomical subjects. This strip will be invaluable not only to speakers wishing to address organisations of any type, but also to those wishing to have a private collection of rocket and space-flight pictures. If a projector is not available, the strip can of course be examined by one of the 35 mm. transparency viewers obtainable at any photographer's.

The cost is 10/- per strip, post free, which works out at the modest sum of 2½d. a picture. Orders should be sent to the Secretary at 12, Bessborough Gardens, S.W.1.

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